SA71

ADVANCED NSTS PROPULSION SYSTEM

VERIFICATION STUDY

FINAL REPORT

July 31, 1989

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Rockwell International
Space Transportation
Systems Division
Huntsville Operations

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STUDY LEADER: Jim Yeager

PREPARED BY: Charles Wood, with contributions from Jim Yeager, Virgil Schaffer, Jim Williams, Mike Hayes, George Nakamura, Dale Wong, and Roland Griner. Advisors Ted Kaser, Tom Baggette, Robbie Robertson, Sammy Lee, and Ed Mosbrook.

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APPROVED BY:

H. R. Wiener
Chief Engineer

J. K. Williams
Program Manager

Rockwell International

Space Transportation
Systems Division
Huntsville Operations

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LIST OF ACRONYMS

AHP	analytical hierarchy process
APU	auxiliary power unit
ARMA	accumulator reservoir manifold assembly
ASI	augmented spark igniter
CCV	chamber coolant valve
CDDT	countdown demonstration test
CRES	corrosion resistant steel
CSM	command and service module (Apollo)
DAC	Douglas Aircraft Company
DDT&E	design, development, test, and engineering
ECO	engine cutoff sensors (propellant depletion)
EIU	engine interface unit
ERD	electrical reference designator
ET	external tank
FASCOS	flight acceleration safety cutoff system
FCHL	Flight Control Hydraulic Laboratory
FMOF	first manned orbital flight
FRF	flight readiness firing
FRFs	flight readiness firings
GG	gas generator
GGOT	gas generator overtemperature cutoff
GLV	Gemini launch vehicle
GPM	gallons per minute
GSE	ground support equipment
HFA	high frequency accelerometer
HPFTP	high pressure fuel turbopump
HPOTP	high pressure oxidizer turbopump
ICD	interface control document
IECO	inboard engine cutoff
LFA	low frequency accelerometer
LPOG	liquid propellant gas generator
LPOTP	low pressure oxidizer turbopump
LPS	launch processing system
MCC	main combustion chamber
MDM	multiplexer/demultiplexer
MDS	malfunction detection system
MDTCPS	malfunction detection thrust chamber pressure switch
MEC	main engine controller
MFV	main fuel valve
MFVs	main fuel valves
MPS	main propulsion system
MPT	main propulsion test
MPTA	main propulsion test article
MSFC	Marshall Space Flight Center
MTF	Mississippi Test Facility (now Stennis Space Center)
MVGVT	mated vehicle ground vibration test

NASA	National Aeronautics and Space Administration
NPSH	net positive suction head
NPSP	net positive suction pressure
OECO	outboard engine cutoff
OPOV	oxidizer preburner oxidizer valve
OTBV	outboard bleed valve
PMDS	prelaunch malfunction detection system
POGO	propulsion system dynamic interaction with vehicle structure
PSI	pounds force per square inch
PSIA	pounds force per sqaure inch, absolute
PSIG	pounds force per square inch, gage
PSTP	propulsion system test program
PSTP	propulsion system test program (Gemini)
PU	propellant utilization (system)
RESS	redundant engine shutdown system
ROM	rough-order of magnitude
RTLS	return-to-launch site
SAIL	Shuttle Avionics Integrated Laboratory
SATS	Shuttle avionics test set
SCIM	standard cubic inches per minute
SPGG	solid propellant gas generator
SRB	solid rocket booster
SRMs	solid rocket motors
SSC	Stennis Space Center (NSTL, MTF)
SSME	Space Shuttle main engine
SSMEs	Space Shuttle main engines
STSD	Space Transportation Systems Division
T&V	test and verification
TCDS	terminal countdown sequencer
TCPS	thrust chamber pressure switch
TCV	thrust chamber valves
TPS	thermal protection system
TVC	thrust vector control

INTRODUCTION

PURPOSE

The purpose of this study was to determine the merits of propulsion system development testing utilizing the existing data base of technical reports and people and to organize and present available data along with conclusions and recommendations for use by management in the structuring of future vehicle development programs.

SCOPE

The study encompassed a review of all available test reports of propulsion system development testing for the Saturn stages, the Titan stages, and Space Shuttle main propulsion system. The knowledge on propulsion system development and system testing available from specialists and managers was also "tapped" for inclusion. The data from the numerous sources was analyzed and is included in the report along with conclusions and recommendations, program risk involved if propulsion system testing is not conducted, and a verification trade study to select a preferred verification alternative using the analytical hierarchy process (AHP).

AUTHORITY

This study was performed in response to a task from Stennis Space Center to contract NAS8-36700.

DATA GATHERING PROCESS

Static firings that were chosen as having applicability to this study went back approximately 30 years, consequently considerable time was spent locating data for screening. The MSFC Repository and Library, the Redstone Scientific Information Center, the MSFC Technical Information and Services Branch, the University of Alabama-Huntsville library, and a number of individual employees of NASA and the Saturn stage contractors were contacted for leads to Saturn data. Some documents were located in the personal files of individuals, however, it was found that the major part of the documents had either been destroyed or had been sent to the Federal Archives and Records Center in East Point, Georgia. Retrieval from that location proved to be quite lengthy and incomplete. To provide additional data, a survey form requesting data inputs was also distributed to 138 propulsion system specialists.

SUMMARY

The importance of, and necessity for, propulsion system testing has been investigated to provide a data base and recommendations for assisting program managers who may be confronted with propulsion system testing decisions.

Two complementing approaches have been used in the study implementation. First, a survey form and instructions distributed to 138 propulsion system/engine design and test experts throughout the nation requested information relative to propulsion system testing, and secondly, all NASA launch vehicles and some Air Force missile development programs since 1960 were evaluated relative to constructive contributions from propulsion system test programs. Significant findings from these activities, including conclusions and recommendations, are presented.

Program managers and chief engineers, managers of major engineering organizations, and active design and test engineers experienced in both vehicle and engine development were included in the survey. In response to one major survey request, "Summarize your opinion of the role of "all-up" system testing in verification of a new propulsion system prior to first launch," response strongly supported/urged such testing with most considering such testing mandatory. Only one respondent was opposed to system testing. A few expressed qualifications related to manned versus unmanned vehicles and similarity of designs with previous designs. A series of other survey "choice type" questions permitted assessment of the relative importance of cost, schedule, and reliability. As was expected, reliability was easily the winner.

Evaluation of previous vehicle development programs was accomplished by obtaining and reviewing static firing test reports from battleship, all systems vehicle, and in some cases, flight stages acceptance firings. Other data sources were used where noted test reports were not available. For S-IV, documentation was totally unavailable, thus personnel memory was the data source. Programs included in the evaluation were all stages of Saturn I (1959-1965), Saturn IB (1961-1968), Saturn V (1961-1972), and Space Shuttle from NASA's reservoir and Titan ICBM, Gemini Titan, and Space Titan from the Air Force reservoir. These programs represent a wide spectrum of propulsion system hardware — single and multi tanks for each propellant, various propellant combinations including both cryogenic and storable, various engine designs including both single and clustered engine configurations, propulsion systems for booster stages, and space type vehicles with altitude ignition.

Compared with today's state of the art in rocket engine and stage design and testing, the technological base in the late 1950's and early 1960's was lacking in many areas. As a result, the Saturn I, IB, and V programs, initiated at that time, experienced many problems that in hindsight could have been prevented. The Shuttle program and Air Force programs, initiated approximately 10 years later, benefited from these early mistakes. While some problems encountered on Shuttle and late Air Force programs were common with those of earlier programs, many

had not previously been experienced. Similarly, for any program initiated today, Shuttle experiences should be more valuable than prior program experiences. Thus in evaluating data within this report, experiences on Shuttle are emphasized since it represents later technologies.

Included in the report are vehicle configuration descriptions, system operation descriptions, system test objectives, and test accomplishments. Accomplishments are presented based on an evaluation of flight or launch operation failure avoidance. Data is also included on hardware changeout frequency, hazardous leakage, fires, and safing procedures developed during testing. Lessons learned, test program cost considerations, test schedule requirements, and representative future test program requirements are discussed.

Even though Space Shuttle development could capitalize on the data base created by previous programs and many management, design and test personnel involved in Shuttle had been involved in prior programs, Space Shuttle propulsion system testing identified, as reported herein, three discrepancies which could be catastrophic during flight and three during preflight operations. Also identified were 5 additional discrepancies classified as unworkable which would be experienced in flight and 17 during preflight operations. Unworkable implies a mandatory change, although the consequence may not be catastrophic. Other less serious discrepancies were observed and numerous product improvement activities resulted from the program. The above tabulations are conservative inasmuch as most vehicle and engine hardware descriptions are excluded from the above tabulations and these discrepancies are numerous and serious as evident from a study of the included tabular data. A careful review of Space Shuttle discrepancies, cost data suggesting prevention of one vehicle loss or loss of one mission may exceed total test program cost, and test schedule data suggesting serious program delays without propulsion system testing removes all doubts relative to the program benefits of propulsion system testing. This deduction is further emphasized by reviewing the risks associated with individual disciplines c' propulsion system design in the absence of a supporting propulsion system test program. Risks for potential launch delay are high for most disciplines, and risks for catastrophic flight failure and mission loss are high for a number of disciplines.

Discrepancies in operational procedures, inadequacies in the vehicle propulsion systems and hardware, and inadequacies in engine hardware were also numerous for NASA programs prior to Shuttle. Air Force programs experienced difficulties also; however, available information on Air Force programs is incomplete, thus its usefulness relative to the current study is limited. Problems/discrepancies encountered in the earlier NASA programs involved most disciplines. For example, on Saturn I (the clustered tank and engine configuration) the flexible heat shield about the four gimballed engines failed structurally in the severe thermal, acoustic, and dynamic environment, thus requiring extensive development prior to flight. On Saturn V S-IC vehicle, helium bottle retainer nuts in the liquid oxygen tank became loose during firing. Also, the terminal countdown sequence was found to be flawed, thus output of all commands occurred at one time when a malfunction or loss of power supply occurred. On the Saturn V S-II stage numerous switch failures occurred, thus soldered seals were abandoned for

welded seals. Also liquid oxygen tank fill-and-drain valves failed to actuate on command and particulate contamination resulted in unacceptable leakage. On Saturn V S-IVB stage, whose initial start occurred at altitude approximately eleven minutes after liftoff with second start in earth orbit several hours after first burn completion, data was collected relative to engine and feed system "warm up" characteristics after first burn and chilldown characteristics prior to second start. This type information and representative analytical techniques were vital for orbital engine start. A thorough review of these early vehicle propulsion system test findings support the earlier stated conclusion for Shuttle—propulsion system testing provides significant benefits. A similar tabulation, as noted earlier for Space Shuttle, of numbers of problems having potential catastrophic effects, and features which were unworkable is impressive.

Propulsion system testing is expensive, requires extensive time to properly plan and implement, and can conceivably delay initial flight vehicle launch. The test article cost can be affected by the extent of new technology utilization, vehicle complexity and the extent to which vehicle launch facility interface hardware is simulated. Cost data based on a number of programs indicated propulsion system testing activities which prevent loss of only one vehicle may be cost effective. Vehicle programs can differ in purpose, thus risks which programs can accept may vary significantly. Vehicle complexity may vary dependent upon flight mission and also the extent to which advanced technology concepts and hardware designs are used. These various aspects and others have been evaluated and are reported in appropriate text, conclusions, and recommendations. Man rated programs require the highest reliability and are most demanding of propulsion system test programs, while unmanned, expendable programs using state-of-the-art technology are the least demanding.

Trade-offs relative to the method used to collect necessary propulsion system data also are possible. Programs with less demanding needs may forego the formal propulsion system test program for an expanded FRF plus analyses or an expanded FRF, flight test program and analyses. The engineering staff and program manager must realistically evaluate each situation giving proper consideration to parameters such as loss of life potential, cost, schedule, advanced technology involved, design similarity with other programs, test site capability, and other factors. The report text provides propulsion system testing requirement guidance to the maximum extent possible.

Table 3-1 (appendix), for Space Shuttle, lists the hundreds of specific individual technical requirements for which test data was sought and obtained. While these many requirements are of utmost importance, the primary reason for testing is to identify unknowns, many of which result from the interactive environments of the many independently designed systems. <u>Analytical models and personnel insights are incapable of a mature representation of all events</u>.

NASA'S HISTORICAL VERIFICATION PROCESS

The Saturn V vehicle's first launch on November 9, 1967, was almost perfect in spite of the fact that neither the S-IC stage nor the S-II stage had been flight tested. The restart of the S-IVB stage, after a three hour orbital coast, was equally successful on the first attempt.

Books have been and will continue to be written on the success of the Saturn launch vehicle. It is contended that never before nor since the Saturn program has a technical endeavor been so successful. There was not one Saturn mission failure, no goal was written off, and most of all, national pride flourished for a brief time, without grief over loss of life in flight. Table 1 lists Saturn launches and provides information on each.

The Space Shuttle continues to provide a never before achieved flexibility in space operations. The reusable spacecraft offers a large cargo capacity and relatively mild launch environment. These characteristics enable earth orbit and outer space launch capability for a wide variety of payloads previously restricted due to weight, shape, or launch sensitivity.

SATURN STAGES

A very real, much overlooked reason for the tremendous Saturn success was extensive ground testing and retesting—including propulsion system testing. Stage static firings were conducted during early development, during late development, for flight verification, between test flights for anomaly resolution, and finally for checkout prior to each launch.

Starting with the first clustered engine firing in March 1960, NASA's emphasis was on stage static firing or, in other words, propulsion system testing. NASA's approach was first to prove system concepts using heavy-weight tanks and non-flight components where necessary and then to perform development testing to refine the stage and procedures with flight components where possible or practical.

Wet countdown demonstration tests (propellant loadings) were performed at the launch site. S-IB, S-IV, and S-IVB all-systems-test-stages were either never built or never fired; however, much development testing was actually accomplished during flight stage acceptance firings. Each stage flown was static fired or acceptance tested at least once prior to launch.

The propulsion test verification process that evolved for Saturn consisted of:

- Component development test program
- Component qualification test program
- Special subsystem testing

TABLE 1. LIST OF ALL SATURN LAUNCHES*

PROGRAM	LAUNCH VEHICLE	MISSION DESIG	LAUNCH DATE	PAYLOAD	DESCRIPTION	REMARKS
	SA-1	----	10-27-61	Dummy	R&D, test S-1 stage propulsion verify structure & aerodynamics	Objectives achieved
	SA-2	----	04-25-62	Water (95 tons)	R&D, observe water dispersion at high altitude	"Project Highwater" (release 22,900 gal water)
	SA-3	----	11-16-62	Water (95 tons)	R&D, observe water dispersion at high altitude	"Project Highwater" (release 22,900 gal water)
	SA-4	----	03-28-63	Dummy	R&D, demo engine-out capability (in-flt eng cutoff)	Objectives achieved
Saturn I	SA-5	----	01-29-64	Dummy	R&D, 1st flt operation of S-IV second stage	First flt operations of S-IV second stage
	SA-6	----	05-28-64	BP-13	R&D, verify struct & aerodynamic design of Sat-I with Apollo boilerplate	Successful insertion into orbit following premature cutoff of one 1st stage engine
	SA-7	----	09-18-64	BP-15	R&D, demo of LES jettison	Active ST-124 guidance
	SA-8	----	05-25-65	Pegasus 11 BP-[illegible]	Operational, meteoroid experiment near Earth environment	Successful 1st CCSD-built S-1 stage
	SA-9	----	02-16-65	Pegasus 1 BP-16	Operational, meteoroid experiment near Earth environment	Successful
	SA-10	----	07-30-65	Pegasus III BP-9	Operational, meteoroid experiment near Earth environment	Completed Saturn I program 2nd CCSD S-1 stage

* Extracted from the NASA History Series, "Sta to Saturn", Roger E. Bilstein, 1980.

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TABLE 1. LIST OF ALL SATURN LAUNCHES* (Con't)

PROGRAM	LAUNCH VEHICLE	MISSION DESIG	LAUNCH DATE	PAYLOAD	DESCRIPTION	REMARKS
SATURN IB	SA-201	AS-201	02-26-66	CSM-009	R&D, CSM subsys & struct integrity & veh compatibility	Reentry adequacy was demonstrated under Earth orbital conditions.
	SA-202	AS-202	08-25-66	CSM-011	R&D propulsion & entry control by G&N system	Demonstration of entry at 28,500 FPS
	SA-203	AS-203	07-05-66	LH_2 in S-IVB	R&D, control of LH_2 by continuous venting in orbit	Successful (4 orbits)
	SA-204	Apollo 5	01-22-68	LM-1	LM dev, verify ascent & descent prop sys eval LM staging	Successful (4 orbits)
	SA-205	Apollo 7	10-11-68	CSM-101	Operational, first manned CSM operation	163 orbits, off Earth duration 10 days & 20 hrs.
	SA-206	SL-2	05-25-73	CSM-116	First manned launch to the Earth orbiting space station. Repaired damaged solar array wing & deployed parasol	Duration 28 days
	SA-207	SL-3	07-23-73	CSM-117	Second manned launch to the Earth orbiting space station. Solar data, EREP, & biomedical experiments	Duration 59 days
	SA-208	SL-4	11-16-73	CSM-118	Third manned launch to the Earth orbiting space station. Solar data, EREP & biomedical experiments	Duration 60 days Open-ended to 85 days

* Extracted from the NASA History Series, "S to Saturn", Roq . E. Bilstein 80.

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TABLE 1. LIST OF ALL SATURN LAUNCHES* (Con't)

PROGRAM	LAUNCH VEHICLE	MISSION DESIG	LAUNCH DATE	PAYLOAD	DESCRIPTION	REMARKS
	SA-209	ASTP backup	------	CSM-119	Provided SL crew rescue capability until 2/8/74 (splashdown of SA-208)	SL mission successfully completed 2/8/74
SATURN IB	SA-210	ASTP	07-15-75	CSM-111	Conduct manned rendezvous and docking mission with U.S.S.R. (Soyuz)	----
	SA-211	Mission not assigned		----	----	----
	SA-501	Apollo 4	11-09-67	CSM-017 LTA-10R	R&D, launch veh & SC dev Sat veh performance	CM entry at lunar return velocity (three orbits)
	SA-502	Apollo 6	04-04-68	CSM-020 LTA-2R	R&D, demo of S-IC/S-II & S-IVB separation	Eval of EDS closed-loop configuration (three orbits)
	SA-503	Apollo 8	12-21-68	CSM-103 LTA-B	Operational, first manned lunar orbital mission	20 hrs in lunar orbit (10 orbits). Off Earth duration 6 days & 3 hrs
Saturn V	SA-504	Apollo 9	03-03-69	CSM-104 LM-3	First manned CSM/LM oper demo lunar orbit rendezvous in Earth orbit	Off Earth duration 10 days & 1 hr (152 orbits)
	SA-505	Apollo 10	05-18-69	CSM-106 LM-4	First manned CSM/LM oper in cislunar & lunar environment	Simul lunar landing mission 61.6 hrs in lunar orbit (31 orbits). Off Earth duration of 8 days

* Extracted from the NASA History Series, "Stages to Saturn", Roger E. Bilstei 1980.

(M-6.8/SL.3)

TABLE 1. LIST OF ALL SATURN LAUNCHES* (Con't)

PROGRAM	LAUNCH VEHICLE	MISSION DESIG	LAUNCH DATE	PAYLOAD	DESCRIPTION	REMARKS
	SA-506	Apollo 11	07-16-69	CSM-107 LM-5 EASEP	First manned lunar landing mission development EASEP	One EVA 2.5 hrs lunar stay 21.6 hrs. Off Earth duration 8 days & 3.3 hrs.
	SA-507	Apollo 12	11-14-69	CSM-108 LM-6 ALSEP	Second manned lunar landing mission deploy ALSEP. Surveyor III investigation	Two dual EVAs 4 hrs & 3.75 hrs. Off Earth duration 10 days & 4.6 hrs
	SA-508	Apollo 13	04-11-70	CSM-109 LM-7 ALSEP	Mission aborted due to failure of SM oxygen storage sys. S-IVB impact on moon	LM lifeboat mode for lunar flyby & return to Earth. Off Earth duration 5 days & 22.9 hrs.
SATURN V	SA-509	Apollo 14	01-31-71	CSM-110 LM-8 ALSEP	Third manned lunar landing, deploy ALSEP lunar surface stay 33.5 hrs.	Two dual EVAs 4.8 hrs & 4.3 hrs. Off Earth duration 9 days
	SA-510	Apollo 15	07-26-71	CSM-112 LM-10 ALSEP LRV-1	Fourth manned lunar landing, deploy ALSEP 3 traverses with LRV-1 6.5 hrs-7.2 hrs-4.8 hrs.	LRV traverses 27.9 km. Off Earth duration 12 days & 7.2 hrs
	SA-511	Apollo 16	04-16-72	CSM-113 LM-11 LRV-2 UV-photo	Fifth manned lunar landing deploy ALSEP-UV camera 3 traverses with LRV-2 7.2 hrs - 7.4 hrs - 5.6 hrs	LRV traverses 26.9 km. Off Earth duration 11 days & 2 hrs
	SA-512	Apollo 17	12-06-72	CSM-114 LM-12 LRV-3, ALSEP & su[illegible]e expr.	Sixth manned lunar landing 3 traverses with LRV-3 7.2 hrs - 7.6 hrs - 7.3 hrs.	LRV traverses distance 35.7 km

* Extracted from the NASA History Series, "Stage[illegible]turn", Roger E. Bilstei[illegible] 1980.

(M-6.8/SL.4)

TABLE 1. LIST OF ALL SATURN LAUNCHES* (Concluded)

PROGRAM	LAUNCH VEHICLE	MISSION DESIG	LAUNCH DATE	PAYLOAD	DESCRIPTION	REMARKS
SATURN V	SA-513	SL-1	05-14-73	Multidocking Adpt. ATM, Workshop Module Airlock	Unmanned launch placed space station in a circular Earth orbit 433 km	Manned logistics: launches SL-2, SL-3, & SL-4
	SA-514	Mission not assigned		----	----	----
	SA-515	Mission not assigned		----	----	----

* Extracted from the NASA History Series, "Sta to Saturn", Roger E. Bilstein, 1980.

(M-6.8/SL.5)

- Propulsion system (stage) development testing
- Stage acceptance firings
- Vehicle flight test program
- Additional stage static testing for continued development, for resolution of in-flight anomalies, and for increased reliability.

Structural test stages, dynamic test stages, and dummy "facilities check" stages were fabricated and tested in addition to propulsion system test articles. The schedule permitted S-I and S-IB flight tests with dummy upper stages but this luxury was not available for the S-IC or S-II stages.

SPACE SHUTTLE MAIN PROPULSION SYSTEM

NASA's Space Shuttle MPS testing program, although abbreviated, was comparable to the previous Saturn program. Three non-firing tests and twelve combination development/verification firings met planned pre-test objectives. The main propulsion test article (MPTA) was flight configuration with a few practical exceptions:

- Non-flight external tank insulation.
- Auxiliary power unit (APU) simulated by ground powered hydraulic system.
- Shuttle avionics test set (SATS) instead of flight computers.
- Truss type load bearing structure forward of the 1307 bulkhead.
- Simulated payload bay purge into aft compartment.
- Some non-flight GSE consoles (however, most GSE was intended to be flight fidelity).
- Non-flight propellant loading configuration (emergency drains added to the external tank).
- Non-flight ground umbilical disconnects

Many MPS components were adapted from Saturn, and thus qualified for main propulsion test (MPT) use by similarity or a delta qualification when necessary. New design components were subjected to rigorous development and qualification testing prior to use on the test article.

MPT static firings were not used to resolve inflight anomalies, in part because no serious anomalies occurred that were not understood. The first launch was April 12, 1981, and the last MPT firing was performed January 17, 1981. However,

propellant loading tests were performed with the MPTA after the flight program started.

Other Development/Verification Testing

Management started questioning the use of checkout firings for stage acceptance late in the Saturn development program, thus no true acceptance firing was planned for the integrated Shuttle configuration. A short duration FRF concept finally evolved for new orbiters, a new launch facility, or after extensive down time. It was later seriously argued that this minimum test was unnecessary even after the FRF for Challenger (OV-099) resulted in detection of a gross hydrogen leak which could have been catastrophic under flight conditions (without purging).

Extensive development testing was performed at KSC with the flight vehicle prior to the first launch to improve propellant loading procedures and to resolve other problems. Due to a thermal protection system (TPS) failure on the external tank and to the desire to increase onboard propellant mass, several unplanned propellant loading tests were conducted both with the MPTA and with the flight vehicle at KSC.

In addition to MPT static firings, structural test articles were built and tested, approach and landing tests were performed, an all-up hydraulic simulator Flight Control Hydraulic Laboratory (FCHL) was developed and used, a mated vehicle ground vibration test (MVGVT) program was conducted, a Shuttle Avionics Integrated Laboratory (SAIL) was and is used, along with the Hydraulic Simulation Laboratory (HSL), for software verification prior to launch, and full scale external tank terminal drain tests were conducted.

EVALUATION OF PROPULSION SYSTEM TEST PROGRAMS

An evaluation of propulsion system test programs which have been conducted previously are discussed in this section. The intent is consolidation and focusing of significant aspects of various programs which may be beneficial to future programs. Major emphasis is placed on the Space Shuttle program since it is currently operational; the design is a benefactor of prior programs, and it represents, as much as any program, current, state-of-the-art technology. Data is also presented for the Saturn program. Further information and data on each program are presented in the various appendices.

SPACE SHUTTLE MAIN PROPULSION SYSTEM

MPTA is the Space Shuttle propulsion system test article. The test program was conducted at NASA's test site in Mississippi, and the program involved NASA and Space Shuttle element contractors: (1) Rockwell International, Space Division (orbiter); (2) Rockwell International, Rocketdyne Division (main engine); (3) Martin Marrietta (external tank); (4) Rockwell International, Space Division (integration and test site operation).

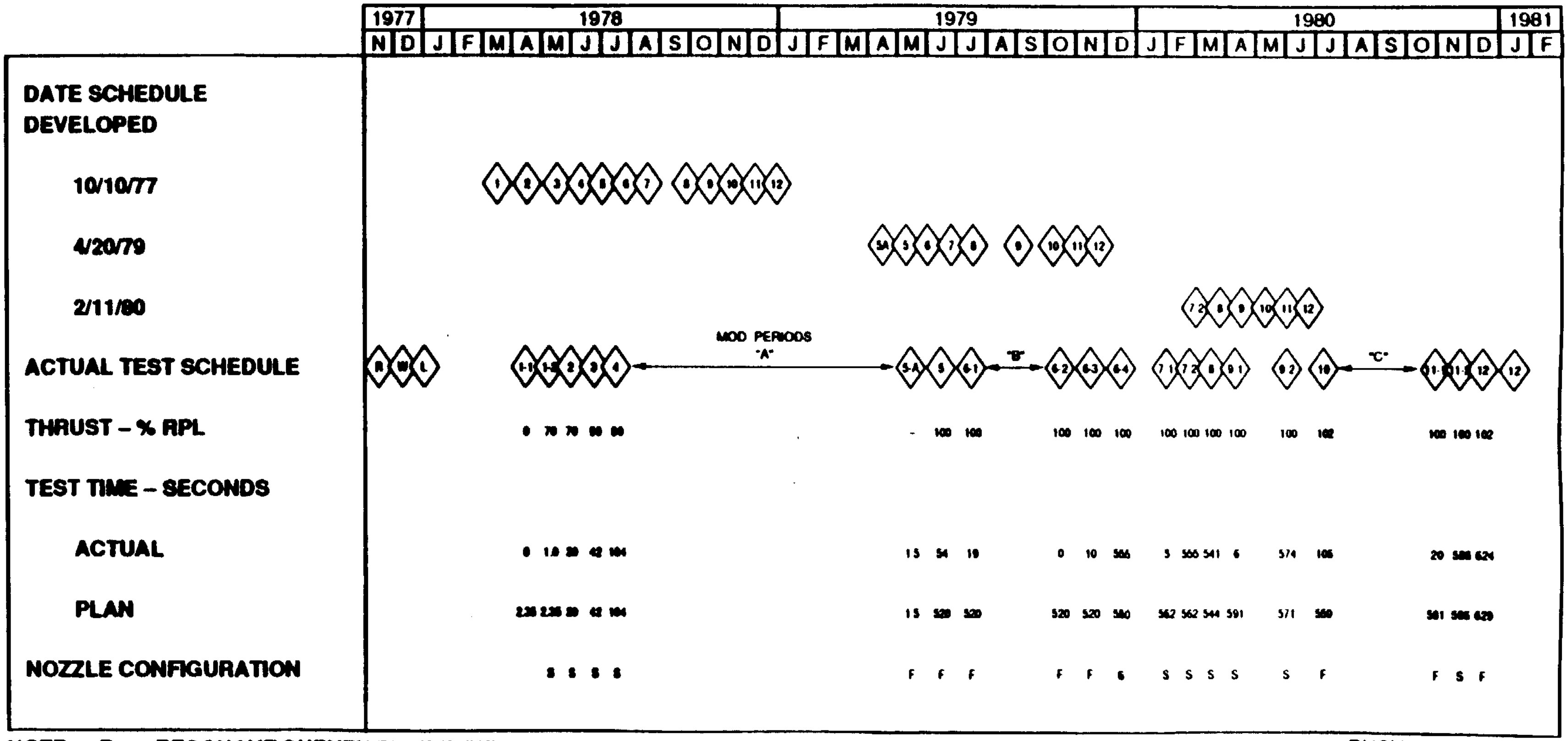
MPTA Test Schedule

Figure 1 contains schedule data for the program. The planned program involved 12 hot firings, 1 propellant loading test, and 2 structural resonant survey tests. The 15 tests plus several attempted tests were completed before the initial Shuttle flight. This program would satisfy all identified requirements presented in Appendix 3 of this report. The schedule for testing established on October 10, 1977, two months prior to test initiation, depicts 12 tests with the last test on December 14, 1978. Test initiation was as planned, but Test 12 was not completed until January 17, 1981—two years later than planned. Later planning schedules, one developed on April 20, 1979, and one developed February 11, 1980, were almost equally in error. The actual schedule indicates 4 tests were necessary to obtain necessary structural and propellant loading data and 20 hot firing attempts were necessary to achieve 12 successful tests.

The actual schedule shows three intervals in which major modifications were necessary. These periods were for updating vehicle and engine hardware and to repair damaged hardware. The total time for these modification periods was 17 months, approximately one half the actual time of the test program.

The following section provides more information on the test program, although important observations are possible from Figure 1. These observations follow:

- The approximate three-week interval between tests reflected in planning activities was correct.



NOTE: R - RESONANT SURVEY TEST (DRY)
W - RESONANT SURVEY TEST (40% O_2)
L - PROPELLANT LOADING
S/F - STUB/FLIGHT NOZZLE CONFIGURATION

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FIGURE 1. MPTA TEST SCHEDULES

- Complexity of the vehicle, facility, and the total operations necessitates recognition of test abort possibilities and inclusion for same in test activity planning.

- Inclusion of time for major modification/repairs is necessary for realistic scheduling of test programs for complex vehicle designs.

- Vehicle maturity will not exist in any complex vehicle design, test hardware, or flight hardware until some minimal system testing is performed.

MPTA Events Summary

A summary of some significant events from the propulsion system test program are presented in Table 2; and for purpose of convenience, information on aborted tests and causes of aborted tests have been extracted from Table 2 and presented in Table 3. Also, for convenience major changes made during the three modification periods are shown in Table 4.

The features of Table 2 may be considered as a measure of vehicle maturity or lack thereof. Table 2 lists all tests, identifies those which were aborted and why aborted, when aborted, those which experienced propellant leakage either internally or externally, those experiencing fires, those for which fire indicators were activated, those for which installed safety enhancement features of internal gaseous nitrogen purges (5,000 and/or 30,000 Scfm) and internal/external water systems were activated and other pertinent data. To assist in comprehending the material in Table 2, the following observations of a statistical nature are presented:

- There were 20 hot firing attempts.

- Two of the twenty attempts did not result in engine ignition. These failures are classified as vehicle failures. Tests 1-001 and 6-02.

- Seven of the twenty attempts were hot firing test aborts. All seven were engine related.

- Hot firing aborts were distributed throughout the 20 firing attempts, although frequency of occurrence decreased after tests 6-03.

- Twenty-six terminal counts were required. Fourteen were required during the first six firing attempts. After test 5A, only one per test was required, although frequent, simple "work arounds" were required.

- Hydrogen leakage within the aft compartment occurred on 12 tests. Two tests experienced high leaks—engine hardware failures—Et/orbiter 17-inch disconnect was a frequent but relatively moderate "leaker."

- The emergency 5,000 SCFM (5K) nitrogen safing purge (ground test system) within the aft compartment was used for 11 tests to enhance safe operation.

Table 2. MPTA Testing Summary

Firing No.	Aborted Normal	Duration Seconds	Terminal Count Attempts	Propellant Leakage	Aft Compartment 5K N2 Purge	Aft Compartment 30K N2 Purge	Fire	Internal Water	External Water	Fire Indication	Remarks
Resonant (dry)		n/a	n/a	n/a							Completed per plan.
Resonant 40% O2	A	n/a	n/a								SATS and SSME communication failed
Resonant 40% O2		n/a	n/a								Completed per plan. O2 pressure spike during draining - hazardous
LO2/LH2 tanking		n/a	n/a								Completed per plan. Many problems/ changes/actions involving safety/ operations - see separate tabulation
SF-001	A	0 vs. 1.35	5								Redline violations; instrumentation drifting/failures - excessive interlocks. Several pages of actions necessary
SF-002	A	1.0 vs. 1.35	1		x		Shutdown fire ball		X	To umbilical	To umbilical water failed (solenoid problem). Test aborted by cross wiring of thermocouple; prevalve switch indicated open when cold; insulation removed from microswitch post test; actuator replaced post test; H2 burnoff problems. Several pages of actions.
SF-02		20 vs. 20	1	Large repaired facility leak. Aft compartment 0.3% H2.							70% RPL thrust.
SF-03		42 vs. 42	3	Small, repaired facility leak. Aft compartment during detanking	Detanking						LH2 recirculation valve open indication necessitated recycles; s.w. bypassed. 90% RPL thrust. .45% H2 concentration. Much instrumentation failures; some component failures. Two H2 burners failed, air/H2 crosswired. High point bleed back pressure reduction. New procedures.
SF-04		104 vs. 104	1								90% RPL thrust
10 MONTH DOWN PERIOD											Many hardware/software changes

Table 2. MPTA Testing Summary (Continued)

Firing No.	Aborted Normal	Duration Seconds	Terminal Count Attempts	Propellant Leakage	Aft Compartment 5K N2 Purge	Aft Compartment 30K N2 Purge	Fire	Internal Water	External Water	Fire Indication	Remarks
SF-5A		1.5 vs. 1.5	3	Aft compartment above 17" ET/ Orbiter disconnect			Extensive burning at shutdown		x		Engine 3 pump inlet overpressure caused recycle/workaround. Faulty circuit prevented recirculation valve from closing. Heat shield water spray used - no damage. Possible MFV #2 leak. Post test faulty engine nozzle weld required changeout. .5% H2 concentration post detanking.
SF-5	A	54 vs. 555	1								Faulty accelometers on two engine HPOPs terminated test.
SF-6-01	A	19 vs. 520	1	Massive aft compartment H2 leak	x	x	Vehicle base and exterior (hugh)		x all sources	x also internal (false)	Engine 1 MFV failed, dumping massive H2 quantities in aft compartment. Rapid pressure rise/structural failure of heat shield and structure. No internal fire damage. Engine 1 HPFTP turbine discharge temperature spike cut test. Simultaneous failure of both channels A and B of Engine1 controller; pneumatic shutdown. 150 seconds post cutoff, emergency H2 drain commenced. No.1 H2 prevalve closure impossible until To +6.5 minutes. Many software changes resulted from this test to enhance operations and safety.
3 1/2 MONTH DOWN PERIOD											Hardware/electrical repair, hardware/software changeout
SF-6-02	A	0 vs. 555	0	Aft compartment indication	x						Fast response H2 gas detector redline exceeded. Serious O2 loading problems. Special test necessary - potential O2 geyser of concern.
SF-6-03	A	10 vs. 555	1		x		x external		17" disconnect and vehicle base	17" ET/ Orbiter disconnect. Vehicle base	Engine 3 HPOTP secondary seal cavity pressure exceeded redline cut test. Sparks from Engine 1 at shutdown, thus extensive internal Engine 1 damage. H2 fire in vehicle base and half of vehicle length. Extensive instrumentation damage. ET insulation char over large area. Engine 1, HPOTP 3, and all 3 flight nozzles replaced.

Table 2. MPTA Testing Summary (Continued)

Firing No.	Aborted Normal	Duration Seconds	Terminal Count Attempts	Propellant Leakage	Aft Compartment 5K N2 Purge	Aft Compartment 30K N2 Purge	Fire	Internal Water	External Water	Fire Indication	Remarks
SF-6-04		550 vs. 543	1	Aft compartment. 2000 PPM H2			x post firing vehicle base		Heat shield water spray cycled on/off after cut-off for ~10 minutes.		
SF-7-01	A	5 vs 550	1								Outboard fill/drain valve stuck open. Test delayed 2 days for replacing. Auxiliary drain valve to drain tank. Valve cycled, LH2 tank spiked to 23 psi (establishing tank pressure cycle) when auxiliary drain valve operation verified. Inspection showed incomplete closure due to level sensor wiring. Engine 3 HPOTP turbine discharge temperature cut test (incomplete ignition of O2 preburner). LO2 tank negative pressure created during draining. Tank inspection and procedure change required.
SF-7-02		550 vs. 550	1				x after shut-off, vehicle base				Engine 2 liftoff seal H2 leak post cutoff. No damage. O2 auxiliary drain valve closed but microswitch would not operate - valve changeout post test. Both facility O2/H2 line leaks - corrected. Many hardware changes from SF-7-01
SF-8		550 vs. 550	1	Aft compartment O2/H2	x cycled on/off						Small leaks; safety maintained by cycling N2 purge. Extensive LN2 formed on aft compartment poorly insulated H2 surfaces. Engine 2 HPFTP replaced post test.
SF-9-01	A	6 vs. 569	1	Aft compartment during initial hot firing and draining	x during drain back		x 17" dis-connect during drain back		x 17" discon-nect during drain	17" ET/orbiter disconnect	Engine 2 HPFTP turbine discharge temperature cut test. LN2 on some cold lines.
SF-9-02		578 vs. 574	1	Loading and firing	x loading						Firing 1200 to 2000 PPM in aft compartment. 5K purge used twice, 1 1/2 minutes each, during fast fill. When on H2 approached zero.

Table 2. MPTA Testing Summary (Concluded)

Firing No.	Aborted Normal	Duration Seconds	Terminal Count Attempts	Propellant Leakage	Aft Compartment 5K N2 Purge	Aft Compartment 30K N2 Purge	Fire	Internal Water	External Water	Fire Indication	Remarks
SF-10-01	A	100 vs. 550	1	Aft compartment. 40 sec. - small H2. 97 sec. - hugh H2	x	x	x within aft compart-ment and vehicle base		x	x	Fire indicator terminated test. H2 preburner burn through. Hot gas dumped in compartment - fire. Also, pressure spike in compartment but no damage. Walter Kidde and IR system detected fire - but Engine 2 sensors detected fire on Engine 3. Hardware, electrical harness damage from fire. LN2 on HPFTP inlet instrumentation.
FOUR MONTH DOWN PERIOD											
SF-11-01	A	20 vs. 574	1	Aft compartment - H2 prefiring/firing. Base fire - post firing.	x	x	x vehicle base				Compartment H2 - 0.5% at To +9.5; 2.0% at To +19.5 sec. Nozzle hole 22 tubes wide by 5" long.
SF-11-02		587 vs. 587	1	Aft compartment H2	x						Main stage hot gas leak. H2 = 0.34% prior to 5K purge. Sensor line connection was source.
SF-12		629 vs. 629	1	Aft compartment - H2 during firing. 300 PPM	x						Free H2 burners failed to provide adequate burnoff for Engine 1. Additional burner added at KSC. Only test without ET anti-geyser line - O2 feedline geyser occurred at transition from chill to slow fill. No damage. Many changes recommended.

Table 3. MPTA Test Abort Summary

TEST NUMBER	TEST DATE	ACTUAL TEST DURATION (sec.)	PLANNED TEST DURATION (sec.)	ABORT SOURCE	DISCUSSION, ABORT CAUSE
1-001	04/11/78	0	2.35	SSME and vehicle	Instrumentation failures. Redline violations.
5	06/12/79	54	520	SSME	Faulty engine pump accelerometers.
6-1	07/02/79	19	520	SSME	Engine main fuel valve cracked and vehicle structural damage resulted. External fire.
6-2	11/24/79	0	520	Vehicle	Aft compartment hazardous gas system faulty reading.
6-3	11/04/79	10	520	SSME	Excessive HPOTP secondary seal cavity pressure cut test. Engine steerhorn (H2 line) ruptured and engine went O2 rich.
7-1	02/01/80	5	562	SSME	HPOTP turbine discharge temperature spike cut test
9-1	04/16/80	6	591	SSME	High HPFTP turbine discharge temperature indication.
10-1	07/12/80	106	550	SSME	Engine H2 preburner burn through resulted in aft compartment fire.
11-1	11/03/80	20	581	SSME	High HPFTP discharge temperature cut test. Nozzle aft manifold failed structurally.

Table 4. Test Hardware Modification Definition

MODIFICATION PERIOD	TIME INTERVAL		PROGRAM DRIVER	DISCUSSION/ CHANGE IDENTIFICATION
	POST TEST	DATE		
"A"	4	07/07/78 to 04/04/79	SSME	Engine pumps capable of RPL operation required. O2 flow meter, HPOTP development, engine main O2 valve vibration problems and SSME 2001 fire complicated development, and extended testing delay. Stub to flight nozzle change.
"B"	6-1	07/02/79 to 10/24/79	SSME	Engine main fuel valve cracked. H2 expelled in large quantities but no internal fire. Investigation necessary plus repair of extensively damaged vehicle.
"C"	10	07/12/80 to 11/13/80	SSME	Compartment fire due to H2 preburner burn through.

- Similarly, the emergency 30K purge was used on three tests.
- Fires occurred for nine firings—one huge external fire, three major external fires with one also internal to the aft compartment. Eight of the nine fires resulted from engine discrepancies.
- Four of the fires near the vehicle base were typical of main fuel valve leaks through the engine after shutdown.
- The one aft compartment fire resulted in extensive hardware damage.
- Two of the external fires produced significant damage to the vehicle and facility—particularly instrumentation.
- Hydrogen gas responsible for the huge external fire originated within the aft compartment and raised compartment pressure sufficient for structural failure of the vehicle heat shield and aft structure.
- Water was used external to the vehicle to suppress fires on eight tests.
- Water was not used within the aft compartment to suppress the single internal fire.
- Fire detection systems identified five fires.

The above observations do not address aspects of technical design which must be verified but attempts to focus attention on the complexity of the total activity and the risk involved in propulsion system development and verification for space and military vehicles. While complexity is apparent from Table 2, it is further demonstrated by Table 5 which is selected action items developed after the MPTA initial propellant loading test and the attempted/accomplished first static firing. Similar actions were necessary for subsequent tests; however, action items numbers and consequences of issues/actions addressed began decreasing after several tests had been conducted.

Risks involved in propulsion system development and verification is not represented completely by Table 2. Another important aspect is associated with specific technical requirements–the real objective of testing–to develop/demonstrate maturity, performance, and functional capability of the vehicle, facility, and GSE, and to demonstrate functional compatibility of the three elements. Detailed test requirements to accomplish these purposes are included in Appendix 3. This test requirement list contained approximately 25 percent fewer entries when initial testing commenced. These "add on" test requirements resulted from oversight, failure to understand functional aspects properly, and hardware changes. Testing identified oversights which could have serious consequences under differing circumstances later in the development program. Five examples follow:

Table 5 *Main Propulsion Test Program - Test Site Edited Action List - Post Propellant LoadingTest (12/21/77)*

ITEM NO	ACTION REQUIRED
1	Review LH_2 high point feedline liquid level sensor operation during propellant load test and repair. Review the need for redundancy
2	Correct channelization of LH_2 tank liquid level sensors (20% to 80% and 98% to 100%)
3	Troubleshoot and isolate the noise problem associated with the EIU SATS interface which resulted in a SATS fail condition
4	Determine the cause of tripped circuit breaker for LH_2 recirculation pump #2
5	Determine cause for the LH_2 disconnect purge
6	Determine requirements for a backup GHe injection system for the LOX antigeyser system operation
7	Review the system requirements and re-activate MEC backup power
8	Provide necessary engineering to connect the KSC hazardous gas detection system to the LH_2 vent system
9	Perform engineering to insulate the RTLS dump line
10	Provide necessary analysis and engineering to correct the LH_2 high point bleed system
11	Perform special cryogenic test of LOX and LH_2 auxiliary dump valves using a solenoid actuated system as well as the existing pressure control actuation system and recommend system changes required to improve valve performance
12	Review the need for changing the 1/8" sample line on the bottom of the LH_2 tank to a 1/4" or larger line to facilitate tank sampling and verification of purge procedures
13	Provide a change to SATS software to preclude opening engine LOX or LH_2 main valves with pressure in feed ducts to prevent potential spinning of engine flowmeters and low pressure pumps
14	Review the need for MEC cooling during the approximate eight-hour period from set-up of pneumatic equipment until start of countdown operations
15	Redesign LOX facility system to provide a safer operational approach to LOX off-loading under existing constraints for the LOX fill and drain system
16	Review and revise LH_2 system purge procedures to effect a better and faster operational method of system inerting, both pre- and post-test
17	Review the adequacy of the fire detection sensors in the vehicle aft compartment
18	Determine the cause and corrective action for the LH_2 prevalve dual indication on engine #2

Table 5 *Main Propulsion Test Program - Test Site Edited Action List (Continued) - Post Propellant LoadingTest (12/21/77)*

ITEM NO.	ACTION REQUIRED
19	Review recent facility system valve failures and determine corrective action
20	Perform a complete design review of SATS, IOPS, EIU, interconnecting cabling, and associated software
21	Troubleshoot the lower vehicle LH_2 capacitence probe to determine the cause of data loss
22	Determine operational changes required to prevent LOX/LH_2 ullage pressure overshoot during terminal count sequence
23	Troubleshoot LOX antigeyser point level sensor measurements
24	Verify that all shipping covers and blanking plates have been removed prior to static firing
25	Evaluate drift in pressure measurements which occurred during propellant load test
26	Data evaluation improvements are needed. Evaluate
27	Automatic propellant replenish system requires modification prior to static firing
	Note: Twenty-eight of forty-eight included

Table 5 Main Propulsion Test Program - Test Site Edited Action List (Continued) - Post Test 1-001 (04/11/78)

ITEM NO	ACTION REQUIRED
1	Redline Review Team - Assess all redline parameters with respect to need and limits. Work with Interlocks Working Group
2	Instrumentation Calibration - Review the method presently used for calibration of critical measurements with respect to accuracy and cryogenic effect
3	Interlocks SATS/MEC - Review the current Sequence and Interlocks Document with the idea of eliminating as many as possible. Emphasis should be on eliminating those analog functions which can be verified visually through other recording and monitoring techniques. Also, determine those software changes required such that they can be coordinated and implemented rapidly
4	Auto Sequence Data Review - Concentrate on the evaluation of the terminal count sequence -540 seconds through T-0 to identify all auto sequence problems
5	Review all Static Firing Setup, Countdown, and Securing Procedures to determine the clarity of instructions, correct labeling of control panels, the inclusion of adequate warning/caution notes, complete/correct instructions for countdown roll-back and recycle operations. Pay particular attention to manual functions not in rollback sequence. Review format for clarity and sequential understanding. Review emergency procedures for setup and presentation for rapid referral when required.
6	Examine LH2 Disconnect Purge ICD values with respect to flow reduction and change hardware to accommodate
7	Determine fix associated with LOX/LH2 tanks overpressure condition reported during test.
8	Review rollback procedure with respect to ET pressurization to reduce number of tank pressure cycles.
9	Review LH2/LOX tank dew point and gas quality requirements with respect to reducing the requirement to that which is necessary rather than that desired. Review purge procedures.
10	Review LOX Chilldown Procedure to determine what can be done to chill down faster with 3,500 GPM orifice installed.
11	Review drop pan arrangement and size and change as required to accommodate the liquid air/nitrogen accumulated during the test.
12	Review procedures/plans with respect to static firing turnaround in case of an abort prior to engine start to minimize work performed prior to the next test. This should include data review, barge turnaround, and retest activity.
13	Review the potential increase in the LH2 tank pressure to off-load at an increased rate.
14	Review the possibility of re-establsihing propellant topping when a hold or a rollback.
	Note: Fourteen of thirty-eight included

Table 5 Main Propulsion Test Program - Test Site Edited Action List (Continued) - Post Test 1-002

ITEM NO	ACTION REQUIRED
1	Conduct a complete review of every redline measurement with respect to the sensor used, how it is transmitted and signal conditioned, how it is recorded, and the method of checkout end-to-end. Include in this review the upper and lower limits of each measurement.
2	Prepare a plan for and complete the troubleshooting associated with SATS/EIU #2 problem.
3	Review the entire software verification program to determine whether or not it si complete. Identify shortcomings and method to correct.
4	Re-evaluate the LH_2 manifold temperature measurement with respect to the probe used versus what is required to produce a valid reading.
5	Investigate the problem of warm LOX and determine changes in loading procedure, hardware design, etc., in order to provide colder LOX at ET/Orbiter interface.
6	Change OCP-M0005 to positively start the LH_2 recirculation pumps prior to termination of fast fill.
7	Determine system changes, i.e., hardware/software required to prevent pre-pressure overshoot and vent valve cracking prior to the next static firing.
8	Correct operator panel legends between Firex water console and fire detect console for compatibility in order to eliminate confusion when Firex water is required. Review all other panels for similar problems.
9	Investigate the need for redundancy in the Free Hydrogen Burnoff temperature sensors and igniters to prevent potential countdown delays.
10	Determine whether or not the fire ball seen at test termination is normal and, if so, determine the changes necessary to minimize the effect such as adding a purge, leaving engine water on, etc.
11	Establish a team of systems engineers to completely review all Test Profile Tables.
12	Review the instrumentation on the Orbiter feed system with respect to adequacy under dynamic conditions while firing and/or POGO pulsing. Incorporate resonance survey data into review.
13	Perform measurement of ET forward support bolt to determine pre-load relaxation and determine whether retorquing is required.
14	Review the response of the 98% point level sensor in both the LH_2 and LOX systems wit respect to lengthening the time constant for fast fill termination to enhance propellant loading time.
15	Determine the hardware and software required to improve the 60KB system to provide all three SSME parameters simultaneously and alternate page selection from at least two users.
16	Investigate the discoloration of the LH_2 GSE fill and drain flex line with respect to damage. Also, determine if additional heat protection is required in umbilical areas.

- Main fuel valve structural failure and associated release of large hydrogen quantities. Design/manufacturing methods were changed.

- Fuel preburner burn through and associated release of combustion products. Design changes made.

- Automated prevalve closure was impossible under some failure conditions. Manual closure was delayed under some failure conditions until prevalve benefits were seriously compromised—To +6.5 minutes before closure was possible. Control computer/engine controller software changes necessary.

- Faulty procedure for unloading oxygen from ET/orbiter created severe pressure surges within facility hardware.

- Necessity for and location of launch facility igniters to burn engine released raw hydrogen at engine start. KSC established requirements for the igniter system.

Technical accomplishments from propulsion system development testing of Space Shuttle to satisfy the above stated general objectives and the specific technical requirements in Appendix 3 are presented and discussed in Appendix 3. Each accomplishment has been assigned to one of four classifications: (1) potentially catastrophic; (2) unworkable, thus unacceptable without change; (3) workable - modification needed and anticipated/planned; and (4) improvement of some type. Also, each accomplishment is assessed relative to the time of events, detection or likely occurrence—flight or preflight. Table 6 is a summary tabulation of accomplishment as classified. Twenty-eight accomplishments are classified to be catastrophic or unworkable, with three of six catastrophic events related to flight. Three of twenty-two unworkable events are also related to flight. Twenty-eight of the forty total events are preflight related. The three potential catastrophic events for flight occurrence are: (1) structural failure of the main engine steer horn and resultant dumping of large quantities of hydrogen in the vehicle base, (2) the marginal stability characteristics of the ET/Orbiter 17-inch oxygen quick disconnect valve, and (3) the oxygen low level cutoff system inadequacies. While catastrophic and unworkable preflight accomplishments may be viewed to be less serious; they, in fact, are equally serious, as vehicle launch cannot occur and extensive time and testing activity are necessary to satisfactorily resolve them.

Table 6. ***Reported Propulsion System Testing Accomplishments Classified by Consequence and Time Phased — MPTA***

Stage	Catastrophe		Unworkable		Workable Mod. Expected		Improvement		Total Per Stage
	Flight	Preflight	Flight	Preflight	Flight	Preflight	Flight	Preflight	
Shuttle	3	3	5	17	3	6	1	2	40

The preceding discussion attempts to put in perspective the requirement for propulsion system testing, the magnitude of the task, and the complexity and

associated risk in developing and verifying a propulsion system for space vehicles by using actual data from a modern day, current operational vehicle. Maturity is a prerequisite for a successful flight program, thus any new development/verification program may be expected to encounter similar experiences. Conclusions to assist in subsequent program developments are:

- Propulsion system complexity precludes total dependency on analytical methods for establishing flight worthiness.
- Propulsion system maturity results from rigorous system testing.
- Rigorous testing involves significant numbers of system tests and long duration tests.
- High risk to test vehicle/facility are associated with the testing.
- Flight schedules established for programs without adequate propulsion system testing may be relatively meaningless.
- Delay of propulsion system testing until vehicle/engine hardware achieve some minimal maturity may be beneficial for reducing risk, cost, and test time. Launch site hardware and software requirements must be properly considered.
- Propulsion system testing should be considered exceptionally high risk without inclusion of safety enhancement features.

SATURN STAGES

Space Shuttle used data developed during the Saturn Program for vehicle/facility design, manufacturing, and test operations to the maximum extent possible. The Saturn Program had many unique requirements, thus it is important to review Saturn propulsion system testing experiences to ensure that conclusions presented are complete and correct.

Included in this section are all stages of Saturn I, Saturn IB, and Saturn V. More specifically the S-IV, and S-I stage of Saturn I, S-IVB and S-IB stages of Saturn IB and S-IVB, S-II, and S-IC stages of Saturn V. Available data for these stages are less plentiful than for Space Shuttle, thus the assessment is less complete.

Summary of Findings

Table 7 contains tabular information about static firings for various stages developed. Information relative to stage and engine design for the various vehicles are included in Table 8. The time span for these test programs is from early 1960 through August 1968. Several observations are possible from the two tables:

- Two propellant combinations were used on the various stages—oxygen/hydrogen and oxygen/RP1.

Table 7. Saturn Program Summary

VEHICLE	Test Start Date	Test Complete Date	Initial Acceptance Firing Date	Initial Flight Date	Number of Static Tests	Number of Tests Aborted	Number of Test Stages Destroyed	Number of Flight Stages Acceptance Tested	Number of Flight Stages Destroyed in Ground Test	Number of Flight Vehicles Lost in Flight
Saturn V										
S-IC			02/24/66	11/09/67				All (15)	1 SIC-II 06/29/69	0
All System	04/09/65	08/03/67			15**	5	0			
S-II			12/01/66	11/09/67				All (15)	0	0
Battleship	11/09/64	09/04/68			54**	29**				
All System	04/23/66	05/28/66			9	6	1 05/28/66		0	0
S-IVB			05/26/66	11/09/67				All (15)	1 01/20/67	
Battleship	06/19/65	08/20/65			5***					
Saturn IB										
S-IB			04/01/65	02/26/66				All (12)		
All System	10/26/64	11/09/62			3	0	0		0	0
S-IVB			08/08/65*	02/26/66				All (12)	0	0
Battleship	11/24/64	05/04/65			16***		0			
Saturn I										
S-I			04/29/61	10/27/61				All (10)		
All System	03/28/60	07/17/62			20**	6				
S-IV			08/12/63*	01/29/64				All (6)	0	0
Battleship	08/17/62	05/04/63			27***					
All System	01/24/64	01/24/64			1		1 01/24/64			

* Second attempt
** Early test not included - see text
*** Estimate

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Table 8. Saturn Program Design Information Summary

VEHICLE	Number Flown	Number of Failures	Propellant Combination	Engine Designation	Engine Thrust (K)	Number of Engines	Engine Single/ Multi Start	Altitude Start	Engine Throttle Ability	Engine Reuse	Engine Chamber Pressure (PSI)	Existing or New Design (Engine)	Multi or Single Stage Tankage	Stage Common Bulkhead	Stage/Engine Supplier	Stage Propellant Tank Insulation	Stage Control
Saturn V																	
S-IC	13	0	O2/RP-1	F	1,500	5	1	No	No	No	1100	New	Single	No	NASA/MSFC, Boeing, Rocketdyne	None	4 O.B. engine gimbal
S-II	13	0	O2/H2	J2	200	5	1	Yes	No	No	685	New	Single	Yes	North American, Inc., Rocketdyne	H2 tank external	4 O.B. engine gimbal
S-IVB	13	0	O2/H2	J2	200	1	2	Yes (twice)	No	No	685	New	Single	Yes	Douglas Aircraft, Rocketdyne	H2 tank internal	Engine gimbal
Saturn IB																	
S-IB	10	0	O2/RP-1	H1	200	8	1	No	No	No	550	New, but similar prede-cessor	Multi (502, 4RP-1)	No	NASA/MSFC, Chrysler, Rocketdyne	None	4 O.B. engine gimbal
S-IVB	10	0	O2/H2	J-2	200	1	1	Yes (once)	No	No	685	New	Single	Yes	Douglas Aircraft, Rocketdyne	H2 tank internal	Engine gimbal
Saturn I																	
S-I	10	0	O2/RP-1	H1	165	8	1	No	No		550	New, but similar prede-cessor	Multi (502, 4RP-1)	No	NASA/MSFC, Rocketdyne	None	4 O.B. engine gimbal
S-IV	6	0	O2/H2	RL10	15	6	1	Yes	No	No	300	Existing	Single	Yes	Douglas Aircraft, Pratt and Whitney	H2 tank internal	4 engines gimbal
Space Shuttle		1 (solid rocket failure)	O2/H2	SSME	470	3	1	No	Yes +9; -35 percent	Yes	3000	New	Single	No	Rockwell, STD; Martin Marietta, Rocketdyne	H2 tank external, O2 tank external, ice prevention	Engines gimbal

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- Various stages used either single or multi tanks for each propellant.

- Stages used from one to eight engines.

- Stages with single burn capability and both sea level and altitude start capability are included, as is a stage having duel burn capability—altitude start and earth orbit start.

- All flight stages were acceptance tested prior to flight.

- All stages were successful during flight.

- Two flight stages were destroyed during acceptance testing.

- Two propulsion system development stages were destroyed during testing.

- Original planning for each oxygen/hydrogen stage included both battleship and all system test hardware, whereas each oxygen/RP1 stage included only all systems "equivalent" hardware. S-IVB/V all system stage was diverted to structural testing after structural test stage failure.

- The S-II all systems stage was destroyed early in the test program.

- The S-IVB/V battleship stage testing was very limited, and the stage was shipped to Arnold Engineering Development Center for altitude testing. Many stage requirements were satisfied by S-IVB/IB battleship testing. Note S-IVB/IB was a single burn stage, whereas S-IVB/V was essentially the same design but with dual start capability.

- Numbers of attempted static firings varied significantly for various stages—a sum of 63 for S-II battleship and all systems test article to 15 for S-I and 15 for S-IC. Numbers of tests exclude several ignition tests for each test article and initial testing when less than a full complement of engines were tested. See later text.

- Similarly, the number of aborted tests varied significantly—from 6 for S-I, 5 for S-IC, and 33 for S-II.

- S-IB stage development testing is limited since differences from S-I are not significant.

- Data for S-IV and S-IVB stages are incomplete. Presented firing attempts are believed to be approximately correct, but test aborts and other data are lacking.

<u>Stages Destroyed</u>

Two ground test stages were destroyed during development testing as were two flight stages during acceptance testing. Such stage loss must be prevented in future programs. Lessons learned from these experiences were highly beneficial

to the Shuttle program and should benefit future programs. The S-IVB flight stage was destroyed prior to actual hot firing when an ambient temperature high pressure helium storage sphere located internal to the stage on the thrust structure ruptured. Post rupture failure analysis identified unacceptable welding material had been used in sphere construction. The S-IC flight stage failure experienced a hydraulic fluid fire internal to the stage which resulted in an aborted firing with much propellant remaining onboard. Improperly implemented securing procedures resulted in the oxygen feed system geysering, not once but twice, with resultant feed system structural failures. Approximately 50,000 gallons of oxygen was dumped within the stage. The S-II all systems test stage failed during check out operations. The liquid hydrogen tank ruptured during helium pressurization. Tank pressure was significantly above acceptable limits through a series of human errors. The oxygen tank of the S-IV all systems test stage was overpressured preparatory to a static firing. The stage was destroyed and the facility badly damaged. Redundant tank vent valves failed to relieve—the pilot valves were adversely affected by solid oxygen particles created by extremely cold helium pressurant gas.

Discussion of Development Firing Programs

Information presented in Table 9 provides greater insight into test programs for which data is available. The figure includes such things as: (1) the number of planned and completed full duration firings; (2) the number of tests which were terminated prior to plan because of stage fires, malfunctioning instrumentation which aborted tests, observers inadvertently aborting tests, tests aborted for legitimate reasons; (3) tests conducted early in the respective programs which have been excluded from other numbers presented in this table and in Table 7.

S-IC. It is obvious from Table 9 data that S-IC was an exceedingly successful test program. Necessary test data were obtained from one vehicle configuration, number of tests were reasonable, preparatory testing considering a new test stand and a stage of such enormous thrust were reasonable, no fires developed and gas leaks occurred on only two tests. Instrumentation failures caused no test aborts, although three tests were aborted unnecessarily by human error. These results are explainable: (1) design experience—the stage was designed and manufactured by MSFC who had designed, manufactured, and tested the S-I stage and two earlier military vehicles; (2) the propellants were the same, oxygen/RP1, used on previous vehicles; (3) MSFC designers/managers had worked closely with the rocket engine developer on prior programs and each respected the other and were knowledgeable of work methods.

S-I and S-IB. The S-I stage was also a highly successful program although considerably more testing was performed than for S-IC. The additional testing is easily justified. The S-I program preceded the S-IC program and represented the "basement floor" in learning and technology development. It represented approximately an eight fold increase in total stage thrust for liquid propelled stages. It used, for the first time, engines in a clustered configuration to create stages with "huge" thrust. It incorporated, for the first time, clustered state-of-the-art manufacturable relatively small diameter tanks to store these "huge" propellant

Table 9. Saturn Program Propulsion System Test Summary

VEHICLE	Tests Attempted	Tests Aborted	Full Duration Tests Scheduled	Duration Scheduled Tests Completed	Short Tests Planned	Planned Short Tests Completed	Tests Legitimately Cut	Tests Cut Because of Fire	Tests Cut Because of Instrumentation Problems	Tests Cut Inadvertently	Preparatory Testing: (not shown in other columns)	Remarks
Saturn V												
S-IC												
All System	15	5	4	3	11	7	2	0	0	3	3 single engine firings	O2 leak 1 test, RP1 and hot gas leaks 1 test
S-II												
Battleship	55	28	25	15	30	12	10	5	12	1	4 single engine firings	3 H2 leak fires/ damage small/2 ASI line failure fires
All System	8	5	3	1	5	2		1	4	0	0	Fire, O2 ASI failure
S-IVB												
Battleship	5											
Saturn IB												
S-IB												
All System	3	0	1	1	2	2	0	0	0	0	0	Development testing on S-I program MFV leakage 2 tests
S-IVB												
Battleship	16											
Saturn I												
S-I												
All System	15	6	12	6	3	3	2	2	4	0	8 clustered engine tank demonstration	Fuel leaks/3 tests. MFV leak last test
S-IV												
Battleship	27											
All System	1											

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quantities, and it required large thrust multi-use test stands with water cooled deflectors. This program was truly a learning experience in liquid rocket propulsion. No test or flight stages were lost. Fires were experienced on only two tests, and damage was small. Four of the twenty firings were aborted for instrumentation failures, and two additional tests were aborted for legitimate reasons. It served as the data base for the later S-IB stage which used uprated engines, longer tanks, aerodynamic fins for stability purposes, and other minor changes.

S-IV. While no detail data is available for the S-IV stage, the approximate number of attempted firings of the battleship and all systems stages were 28. This stage used oxygen and hydrogen as propellants and an existing engine with 15,000 pound thrust. No stage had previously flown using these propellants, although General Dynamics was developing a stage using the same propellants and engine. Douglas Aircraft (DAC) had acquired rocket development experience with the ballistic missile Thor which used oxygen/RP-1 propellants, but experience with hydrogen within the country, DAC, and MSFC was limited. This program, similar to S-I, while developing a stage which flew successful on six occasions, served the dual purpose of training and technology state-of-the-art advancement for hydrogen rocket systems.

S-IVB. The Apollo program, at some later date, required stage total impulse and thrust greater than S-IV could provide. The S-IVB stage was developed to meet such requirements. It was used on a two-stage vehicle, Saturn IB, and three-stage vehicle, Saturn V. The stage for Saturn V required two separate burns, one approximately seven minutes after liftoff and the second several hours later from earth orbit. DAC was the S-IVB contractor and again propellants were oxygen/hydrogen. A single, newly-developed engine of 200,000 pound thrust was used instead of six clustered engines for S-IV. A total of approximately 21 static firing attempts were conducted with the S-IVB stage. Again data does not exist to develop more details. S-IVB flight stages on Saturn IB and Saturn V were all successful. Without the oxygen/hydrogen data base developed on S-IV, the S-IVB stage development program would have been less flexible, and more static firings would likely have been required.

S-II. Detail data are available for S-II, the second stage of Saturn V, which also used oxygen/hydrogen propellants. The stage was developed by North American Aviation, who had neither designed, developed, nor flown rocket propelled stages. Extensive training was required as noted previously for other stages. This was a "huge" stage using five 200,000 pound thrust engines, and, as was true for other stages of Saturn IB and V, was man rated. Total firing attempts for battleship and all systems vehicles were 63. There were 33 aborted tests. Sixteen successful full-duration firings were accomplished from twenty-eight attempted full-duration firings. Six tests were terminated because of internal fires—relatively small to moderate damage. Ten tests were terminated for other legitimate reasons, and sixteen tests were terminated because of control instrumentation failures. The number of tests may be excessive relative to other Saturn programs and Space Shuttle, which may be considered comparable in complexity.

Pre-Shuttle stage designs and development activities were complex and involved awesome risks. Trained personnel were not available to the same extent as they were for Space Shuttle development, and technology which virtually did not exist had to be developed in stage and engine disciplines. In spite of these circumstances, the vehicle flight programs were all successful and many "lessons learned" were acquired. These lessons were of great benefit to Space Shuttle development, and the total list from Saturn, Space Shuttle, and from Air Force programs discussed later should benefit future programs.

The major conclusion from these pre-Shuttle programs is that the Apollo Program would have been a disaster without the comprehensive propulsion system test program which was conducted. Experiences from the Saturn Program do not violate any conclusions presented earlier for Space Shuttle. Actually, experiences reinforce presented conclusions/accomplishments.

Many technical accomplishments are presented and discussed in the respective appendices, 4 through 8, for all Saturn stages (S-IV stage of Saturn 1 information is incomplete). These data are presented in the same format as was done previously for Space Shuttle's MPTA. Table 10, a tabular summary of this data presented according to technical and timing classifications, contains issues for each stage judged to be catastrophic and/or unworkable for either flight or preflight. A careful review of individual accomplishments in the respective appendices clearly shows the necessity for extensive propulsion system test programs.

A memorandum entitled "Static Firing Saturn Stages" authored by Daniel H. Driscoll, Jr. on December 6, 1968, is included in Appendix 10. The question at hand was the necessity for continuing to conduct acceptance firings on each Saturn flight stage which had been, and was, standard policy. A thorough review of NASA's Saturn Program results to that date and the Army and Air Force's large rocket programs was conducted and results reported in subject memorandum. The objective of the study differed from the current study objectives, however, data presented relates to the current subject and should benefit management. The memorandum includes data on number of flights, number of failures, stages which been acceptance fired, length of acceptance firings, and some causes of stage failures.

Conclusions drawn from the investigation are that acceptance firings of flight stages had been beneficial for enhancing safe flight and that the data provided little insight relative to the approximate time that acceptance firings could be deleted.

Table 10. Reported Propulsion System Testing Accomplishments Classified by Consequence and Time Phased

Stage	Catastrophe		Unworkable		Workable Mod. Expected		Improvement		Total Per Stage
	Flight	Preflight	Flight	Preflight	Flight	Preflight	Flight	Preflight	
S-IC	4	0	3	3	1	0	1	1	13
S-V	2	0	8	8	2	0	1	0	21
S-IVB	8	0	6	3	0	1	2	0	20
S-I/IB	5	1	4	2	2	0	1	0	15
S-IV*	2	0	3	1	0	0	0	0	6

*Incomplete

LESSONS LEARNED

The NASA programs of Saturn and Space Shuttle have remarkable flight records—60 successful missions and only 1 failure (which was attributed to non-liquid propulsion systems). The non-flight phases of development programs have been less spectacular and problems of almost every conceivable type have been experienced. This is to be expected since basic technology establishment and personnel training was necessary in parallel with hardware and system development for many of the programs. Many of these experiences may not be readily anticipated and yet may be beneficial for future programs. Some of these more significant experiences designated "lessons learned" are presented. Presented "lessons learned" are from NASA's Saturn and Shuttle programs and from the Air Force Titan programs.

Presented "lessons learned" emphasize testing and limited design aspects of propulsion systems. The items listed collectively demonstrate the contribution to successful space flight which ground test programs have made. In spite of the many "lessons learned" from the Saturn Program, the Shuttle Program is also credited with "lessons learned," thus it is reasonable to expect future development programs will likewise contribute their own lists.

1. Verify functional performance of feed system; pressure drop, pressure surges, water hammer response, resonance.

2. Preclude combinations of incompatible commodities in fluid systems. Three basic principals to follow:

   a. Provide absolute separation of incompatible commodities. Isolation valves and check valves are not sufficient.

   b Provide absolute controls during assembly, maintenance, checkout to mitigate migration, e.g., provide dry purges, dry enclosures to preclude moisture intrusion.

   c. Maintain systems under significant cleanliness controls during all operations.

3. Assure procedures and processes are incorporated to achieve contamination control of contamination sensitive systems.

4. Verify facility to launch vehicle interface design and operating procedures in conducting a ground test or firing of a flight propulsion system.

5. Simulate actual environments, conditions, and designs. This usually appears to be impractical, costly, or near impossible, and it may be in some cases.

6. Demonstrate margin during subsystem or system level test.

7. Use system level testing to fully understand the environments required for component qualification tests.

8. Incorporation of a document which establishes guidelines for qualification, verification, and acceptance test of components and systems can be beneficial. The importance of the document would be guidelines to demonstrate margin for workmanship as well as performance purposes. The Air Force accomplishes this purpose through implementation of MIL-STD-1540B.

9. Stainless steel tubing control lines are required. Aluminum control lines fail structurally.

10. Fuel and oxidizer prevalves are absolutely required for static firing and recommended for flight. They must be capable, both structurally and functionally, of closure at full flow conditions in the event of an engine failure.

11. Safe engine shutoff by prevalve closure should be demonstrated in a subsystem test prior to propulsion system testing. Valve closure acceleration characteristics must be carefully monitored.

12. Thermal control of propulsion system components and electrical components is critical to satisfactory functional performance. Compartments containing such hardware must be purged for hazardous gases. Local thermal control of some hardware may be required. Caution is of utmost importance as such events as valve loss of lubricant, lubricity may be compromised, hydraulic fluid may freeze, etc. Such events may have disastrous effects to vehicle/engine performance.

13. A loss of ground power may result in an inability to terminate a firing. An emergency system to terminate a firing is required.

14. Single rocket engine development testing should simulate vehicle feed system start/main stage/shutoff conditions and engine compartment thermal conditions as may be realistically possible. This may reduce engine testing and operational aspects of the total propulsion system.

15. Launch facility to vehicle supply lines for cryogenic propellants (particularly oxygen) must be designed carefully/properly to permit flow diversion and control during chill down to prevent vehicle hardware damage. Procedures must be developed and concurred on by both facility and vehicle operations personnel. Verification at a non-launch site is advisable.

16. Cryogen fluids, principally oxygen, geysering during propellant loading and standby operations both pre- and post-firing can create unacceptable pressures leading to structural failure of the feed system and/or tankage and possible vehicle loss. Procedures must be carefully prepared by

knowledgeable/experienced personnel, procedures properly focused, and personnel trained accordingly. Verification at a non-launch site is advisable.

17. The opening of a valve to a system or partial system containing cryogen liquid propellant which have been "trapped" for some finite time can result in serious hardware damage. Provisions for pressure relief should be incorporated or special procedures adopted to minimize damage.

18. Temperature and pressure instrumentation directly mounted to hardware at cryogenic temperature may be expected to drift, thus the indicated parameters may be significantly in error, and unacceptable as "redline" measurements. Test aborts will result. Remote mounting is necessary.

19. A reliable hazardous gas detection and measuring system is required for stage static firing programs and pre-flight operations prior to launch.

20. Rapid response leak detectors located both internal and external to the vehicle are required for safe operation.

21. A reliable fire detection system is required for static test programs and may be required for launch site operations—subject to evaluation.

22. Fire protection capability and internal protective neutralizing purges are required for static test programs and may be required for launch site operations—subject to evaluation.

23. Specific pre-planned and measured personnel training and demonstration of qualification is a prerequisite for reliable, repeatable success. System failures were not responsible for the destruction of four stages in the Saturn program which have been identified.

24. For hydrogen propelled stages, as a minimum, a ground system to safely dispose of engine discharged, unburned hydrogen as part of start up is required to avoid excessive pressure spikes which may be incompatible with the stage.

25. Steps to assure effective shift change communications and use of only experienced and qualified personnel is required.

26. A process where contractor and government safety can perform spot checks on all hazardous work control documentation and operations is beneficial.

27. Availability of an "off line" stage/facility during the early flight phases of a development program have proven highly beneficial. Data obtained from both cold and hot test of S-IVB at MSFC provided essential data for both the S-IVB and S-II stages. MPTA propellant loading tests likewise supported Shuttle.

28. Recirculation of rocket engine plume low energy gases into the vehicle base may be the predominant heat source for clustered engine vehicles. Heat flux measurements from ground test programs to determine the relative contributions of convective and radiative heating and analytical methods developed to compensate for flight altitudes assist/establish design requirements.

29. The phenomenon of "flow-induced vibrations" was initially experienced and recognized as an unusual event. Research programs identified the actual phenomena and developed guidelines for design. Major hardware losses are probable if improperly implemented.

30. The phenomenon of subcooling a liquid propellant such as oxygen by injecting a gas-like nitrogen or helium was discovered and has been used extensively to thermally manage fluids on rocket systems.

31. The necessity for a pressurant diffuser which gently distributes incoming pressurant gas within the tank ullage to avoid rapid ullage pressure decay is a requirement for effective tank pressurization. Extensive ground testing and analytical model development have been conducted to better understand requirements for pressurization systems.

32. Severe fluid oscillations within heat exchangers as cryogenic liquids are converted to pressurant gas can occur and must be avoided. Test and analytical work provide useful design criteria.

33. A complex launch vehicle may have 1000 or more fluid connectors ranging in size from one-quarter inch in diameter to eighteen inches or greater. A large percentage contain combustible fluids at moderate to high pressure, and a single leak can result in fire and vehicle damage/loss. Early design emphasis is a prerequisite to success but is difficult to achieve for such a non-glamorous discipline.

34. Cryogen insulation development for large vehicle tankage is not straight forward, and each program is confronted with the commonly experienced, as well as unique development problems. Some of the common problems are insulation debonding, cracking, tank surface corrosion, inability to adequately withstand aerodynamically created thermal loads, etc. Conducting thorough development/test programs is essential, as is planning for insulation repair at the launch site after initial exposure to cold temperatures.

35. Liquid nitrogen is formed on surfaces with temperatures below the boiling point. This usually occurs in the closed engine compartment of liquid hydrogen and liquid oxygen propelled stages and is to be avoided. High quality insulation on rocket engines and selected stage components is essential but is most difficult to achieve.

36. Inclusion of design margins in the initial subsystem/system designs provides flexibility to absorb stage requirement changes which frequently occur.

37 System level tests are necessary to verify end-to-end tolerances for timing, limit checks, and remedial actions. Incremental subsystem tests do not adequately provide the system response signature found in integrated systems tests.

38. Design, development, and test verification of a pogo suppression system prior to first launch is mandatory.

39. Vacuum jacketed cryogenic line maintenance procedures and requirements must be developed prior to launch operations.

40. A separate and independent test peculiar propellant tank ullage pressure instrumentation system must be installed for test stand operations.

41. Purge and pneumatic gasses must be dry.

42. Micro-switch operation is unreliable under cryogenic conditions.

RISKS OF CURTAILING ALL-UP PROPULSION SYSTEMS TESTING

The purpose of this section is to discuss the actual risks created by curtailing or eliminating system testing from a propulsion system verification program. The approach is to examine key items, define the present level of understanding, and make a judgement on the risk surrounding each item.

The results of the risk assessment for space vehicles and facilities are presented in Tables 11 and 12. The evaluation considers no formal propulsion system test program within the development program. Results in Table 11 are for a new program using oxygen and hydrogen propellants, tankage of similar size and configuration to Space Shuttle, and a one burn mission with ignition on the launch pad. Table 12 is a qualitative comparison of various vehicle configurations and propellants with the configuration and propellant as shown in Table 11.

EVALUATION - OVERVIEW

For evaluation purposes, the propulsion system has been divided into a number of subsystems/disciplines and each is evaluated. Evaluation parameters and other material included are:

1. Risk - catastrophic failure, mission loss, flight hardware damage, launch complex damage, and launch delay and how these risks may be affected by a 20 second FRF type test

2. MPTA experience identification for each evaluation subsystem/disciplines

3. Impact on components, subsystems, and flight test programs attributable to no propulsion system testing

4. Impact to the required flight measuring program

The evaluation is qualitative. It was performed by personnel having many years of experience in propulsion system design, development testing, and flight support. Considered in the evaluation were analytical capability, including deficiencies for each evaluation criteria, inherent design complexity and influence of interacting systems, benefits derivable from propellant loading tests, design and operational aspects of space vehicles, and other factors. For additional insight to data in Tables 11 and 12, a detailed discussion for each subsystem/discipline is presented in Appendix 2. Risk classes used were extremely high, high, moderate, low, or minor.

A number of observations are possible from Tables 11 and 12, and they follow:

1. Five of the fifteen subsystems/disciplines evaluated have risks high or above for catastrophe or mission loss. The five criteria are "wrong" component verification requirement, hazardous fluid leakage, POGO

Table 11. Propulsion System Program Assessment - No Propulsion System Test Vehicle Shuttle Type Tankage

							PLANNED				POTENTIAL		
EVALUATION CRITERIA	VEHICLE FLIGHT CATASTROPHE RISK	MISSION LOSS RISK	FLIGHT HARDWARE DAMAGE RISK	LAUNCH COMPLEX RISK	LAUNCH DELAY RISK	MPTA TEST EXPERIENCE	DELTA COMPONENT PROGRAM IMPACT	DELTA SUBSYSTEM PROGRAM IMPACT	DELTA FLIGHT INCREASES BEFORE OPERATIONAL	DELTA FLIGHT INSTRUMENTATION INCREASE	DELTA FLIGHT INCREASES BEFORE OPERATIONAL	20 SECOND FRF PROBLEM RESOLUTION REMAINING RISK	REMARKS
"Wrong" Component Verification Required	Very High	Very High	Yes	High	High	Yes	No	No	No	Yes	High	Low	Delta instrumentation for enviornment determined
Instrumentation Failure	Moderate	Moderate	Yes	Moderate	Very High	Yes	No	No	No	No	Low	Minor	FRF time inadequate for all measurements
Hazardous Fluid Leakage	High	High	Yes	High	Very High	Yes	No	No	No	No	Low	Moderate	FRF time provides inadequate exposure
POGO Failure	Moderate	High	Yes	Minor	Minor	No	No	Yes	No	Yes	Minor	Moderate	
Thrust Vector Control Failure	Low	Low	Yes	Minor	Low	No	No	No	No	Yes	Minor	Minor	
Pressurization System Performance	Moderate	High	Yes	Minor	Minor	Yes	No	Yes	No	Yes	Moderate	Moderate	FRF time inadequate
Propellant Mass Uncertainty	Minor	Moderate	Yes	Minor	Very High	Yes, very high	No	Yes, if possible	No	Yes	High	Low	
Propellant Loading Procedures/Operations	No	No	High	High	Very High	Yes, high	No	Yes	No	Yes	No	No benefit	
Clustered Engine Performance	Minor	Minor	Yes	Minor	Minor	No	No	No	No	No	Minor	Minor	
Low Level Cutoff Sensor	Minor	Minor	Yes	No	Moderate	Yes, high	No	No	Yes (unmanned) No (manned)	Yes (unmanned) No (manned)	Yes (unmanned) No (manned)	No benefit	Inflight test on manned flights not feasible
Performance Margin Uncertainty	Minor	High	No	Minor	No	No	No	No	Yes (unmanned) No (manned)	Yes (unmanned) No (manned)	Yes (unmanned) No (manned)	Moderate	Inflight test on manned flights not feasible
Engine/Feed System Chill	Minor	Minor	Yes	Minor	High	Yes, high	No	No	No	Yes	No	Minor	Loading test benefits
Tank Insulation	Minor	Minor	No	Minor	High	Yes	No	No	No	Yes (ground)	No	Minor	Loading test benefits
Hardware Thermal Control	Minor	Minor	No	Minor	High	Yes	No	No	No	Yes	No	No benefit	Loading test benefits
Stored Gas Mass, Loading, Operations	Minor	Minor	Yes	Moderate	Minor	No	No	No	No	No	Minor	Minor	

Table 12 Propulsion System Program Risk Assessment — No Propulsion System Test Vehicle — Non-Shuttle Type Tankage — Risk Relative to Shuttle

EVALUATION CRITERIA	SHUTTLE FLIGHT CATASTROPHIC AND LAUNCH DELAY RISKS	DIFFERENCE FROM SHUTTLE					
		LARGER - SIMILAR L/D TANKS	SIMILAR VOLUME BUT COMMON BULKHEAD	SMALLER VOLUME BUT SIMILAR - ALTITUDE START	SMALLER VOLUME, COMMON BULKHEAD, ALTITUDE START	SMALLER VOLUME, COMMON BULKHEAD, ORBITAL START	SHUTTLE VOLUME, SIMILAR L/D, DIFFERENT PROPELLANTS
Pressurization System Performance	Moderate/ Minor		Higher/	Higher/	Much Higher/	Significantly Higher/Higher	*Higher/*Higher
Propellant Mass Uncertainty	Minor/ Extremely High	⇑ SAME	Higher/	/	Higher/	Much Higher/	*Higher/*
Low Level Cutoff	Minor/ Moderate	AS	/	/	/	/	*Higher/*
Engine/Feed System Chill	Minor/High	SHUTTLE	/	Higher/	Higher/	Significantly Higher/Higher	*Higher/*
Tank Insulation	Minor/High	⇓	Higher/	/	Higher/	Much Higher/	*Higher/*
Hardware Thermal Control	Minor/High		/	Higher/	Higher/	Significantly Higher/Higher	*Higher/*

NOTE: 1. Blank spaces denote same as Shuttle–column 2.
2. *Varies with propellant selection.

failure, pressurization system failure, and propellant performance margin uncertainty.

2. Twelve of the fifteen subsystem/disciplines evaluated may be expected to result in flight hardware damage.

3. Five of the fifteen subsystems/disciplines evaluated are ranked moderate or greater risk to the launch facility. These are "wrong" component verification requirement, instrumentation failure, hazardous fluid leakage, propellant loading procedures, and stored gas operations.

4. One or more twenty-second FRF type tests will reduce risks noted above but will not eliminate risk. The risk of four subsystems/disciplines evaluated remain moderate—hazardous fluid leakage, POGO failure, pressurization system performance, and performance margin uncertainty.

5. The potential for launch delay risk are not affected by FRF type tests. Ten of the fifteen evaluation criteria are rated moderate or higher risk with three being extremely high—instrumentation failure, hazardous fluid leakage, and propellant mass uncertainty.

6. Planned increases in component, subsystem, and flight development testing are relatively unaffected, but the potential for increased development flights prior to attaining operational status is likely for three subsystems/disciplines evaluated—"wrong" component verification requirement, pressurization system performance, and performance margin uncertainty.

7. Flight vehicle instrumentation increases are necessary for eleven of fifteen subsystem/disciplines evaluated.

8. Six of the fifteen subsystems/disciplines evaluated will be influenced additionally by changes in vehicle configuration, propellant type, location of vehicle at time of engine ignition, and numbers of ignitions. The risk increases significantly in some cases. The six affected criteria are: (a) pressurization system performance; (b) propellant mass uncertainty; (c) low level cutoff; (d) engine/feed system chill; (e) tank insulation; and (f) hardware thermal control.

9. A common bulkhead configuration vehicle using oxygen and hydrogen propellants with multiple ignition and ignition in earth orbit would have the greater risk.

In summary, the broad message from the above material is that a vehicle development program without propulsion system testing involves significantly greater risks from engineering and hardware failures which adversely affect program objective accomplishment, safety, and flight schedule predictability. While a short duration static firing at the launch site may reduce engineering and safety risks, the remaining risks are relatively high, and flight schedule predictability is

unaffected. The technology base for Shuttle configured propulsion systems is more advanced than for other vehicle configurations and missions, thus engineering, safety, and schedule risks for other configurations/missions will be greater.

The evaluation under discussion includes structural/mechanical design capability for only a limited number of subsystems/disciplines, yet such capability is critical for accomplishing program goals. Some perception of hardware maturity may be achieved from Table 13 where hardware changeout for the MPTA is tabulated for each test attempt. The tabulation shows many vehicle and engine hardware exchanges during the program. The risks in Tables 11 and 12 are thus further enhanced. Tabulations in previously discussed sections dealing with test scheduling, why tests were aborted, propellant leaks and fires experienced for the MPTA program, and some similar data for earlier programs were also not used in the subject evaluation. The risk which is evident from these earlier discussions adds further credence to the risk presented in Tables 11 and 12.

PROPULSION SYSTEM TEST PROGRAM

This report has shown that available design knowledge and insight are inadequate to ensure acceptable hardware and operational procedures can be developed which satisfy both engineering and safety without propulsion systems test data. Discussions have also clarified the role of short duration FRF type testing for resolving issues of concern. While the FRF is beneficial, it fails to satisfy all technical needs and does not benefit concerns relating to launch delay. The obvious question is then what test program is required to satisfy engineering and safety needs? The time and cost required to conduct such a program are important, thus plan implementation to minimize program schedule and cost impacts are necessary.

A test program to satisfy identified needs has been developed and is presented in Tables 14 and 15. In as much as vehicle configuration and mission profile may significantly impact the required test program, a vehicle and mission have been selected. The test program presented is for a launch type vehicle with ignition on the launch pad. Vehicles having altitude ignition, ignition in orbit, or differing radically in configuration would require a few—two or three—additional tests to address specific issues peculiar to those missions.

The test program presented in Tables 14 and 15 is a minimum program and involves data collection for propulsion, structural, electrical, and computer/software needs. Tests are specified for the vehicle without propellants, with propellants aboard and no engines firing, and with engines firing. The test program identifies nine tests of which seven are hot firings and four of the seven are full duration firings. The program includes test objectives, estimated time in hours required to collect the necessary non-firing data, test objectives related to test number, and other pertinent information.

Table 13. Hardware Replacement and Repair by Test for MPTA

MPTA Test Number	Engine	Pump	Controller	Nozzle	Major Valves	EIU/MOM	Other: Flowmeter/gimbal bearing/POGO/Preburner/L.O. Seal	Control Instrumentation
	← ENGINE →							
Resonant 40% O2								
Resonant 40% O2								
Loading Test								
1-001								
1-002							1	
2								
3							1	
4								
5-A	.1	12			9			
5						1		
6-01				1	9	1		
6-02/3		1			7		1	1
6-04	1		1	3		1		
7-01					1			
7-02					2			
8					2			
9-01		1						
9-02		4				1		
10				3	4	10		
11-01		2			7			
11-02				3				
12				3				
Total	2	20	1	13	41	15	3	1

MPTA Test Number	Prevalve	17" Disconnect	Level Sensors	Fill and Drain Valve	LH2 Recirculation System, Pumps, High Point Bleed Valves	Pressurization Orifices Valves, Sensors	Auxiliary Drain Valve	Hazardous Gas Detection - Fire Detection	Cut-Off Sensors	LH2 Diffuser	Control Instrumentation	Other: Feedline Screens, Loading Orifice, Drain Orifice
	← VEHICLE →											
Resonant 40% O2												
Resonant 40% O2												
Loading Test												
1-001												
1-002	3	2			4			4				1 Screen
2			1									2 Screen
3				line					1	1		1 GO2 Vent
4										1	1	
5-A			4							1		2 Vents
5		1		line		4		1	2		1	
6-01											2	
6-02/3					3			4	1	1		
6-04								2	2			
7-01												
7-02						2			4			
8		1				5					1	
9-01			2						1		1	
9-02	1					1			1		1	
10			2			1						
11-01	6					4			2			
11-02	3	1				6						
12				1								
Total	13	5	9	3	7	23	0	11	13	4	7	6

Note: Hardware changes made prior to designated test number

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Table 14. Propulsion System Test Program Launch Vehicle - Ground Ignition

TEST CLASSIFICATION	OBJECTIVE	MOTIVATION		NO. OF TESTS REQUIRED	LENGTH OF TESTS REQUIRED	PRIMARY/SECONDARY OBJECTIVE	TEST NO. ASSIGNMENT	TEST DURATION ASSIGNMENT/SECONDS	REMARKS
		TECHNICAL	SCHEDULE						
Non-Firing									
	Resonant Model Vibration	X		2	4 hrs each	P	1/2	n/a	Test 1 - dry; Test 2 - O2 only
	Engine/Tank Purge	X		3	1 hr. each	P	2/3/4	n/a	Test 2 - O2 only; Test 3/4 - O2/H2
	Propellant Loading Procedures	X	X	3	1 hr. each	P	2/3/4	n/a	Test 2 - O2 only; Test 3/4 - O2/H2
	Propellant Mass		X	3	3 hrs. each	P	3/4/5	n/a	All have O2/H2
	Engine/Feed System Chill		X	3	1 hr. each	P	2/3/4	n/a	Test 2 - O2 only; 3/4 - O2/H2
	Hardware Thermal	X	X	3	3 hrs. each	P	3/4/5	n/a	
Firing									
	Component Environment Requirement	X		3	2 full duration minimum	P	5/6/7	60/full	One of two with simulated engine loss
	Instrumentation Verification		X	4	2 full duration minimum	P	4/5/6/7	15/60/full	Data also available Test 3
	Hazardous Fluid Leakage	X		3	2 full duration minimum	P	5/6/7	60/full	Data also available Tests 3/4
	Pressurization System Verification	X		4	3 full duration minimum	P	5/6/7/8	60/full	One of three with simulated engine out; one of three with failed control valve
	Propellant Margin Uncertainty	X		3	3 near full duration minimum	P	6/8/9	Full	3 tests with all engines
	POGO Failure	X		4	25% total firing time	S	5/7/8/9	60/full	One test period with engine out; one test period with engine gimballing
	Thrust Vector Control Failure	X		3	15% total firing time	S	5/6/8	60/full	
	Clustered Engine Performance	X			2 short/1 long	S	3/4/6	2.5/15/full	
	Low Level Cutoff	X		3	Full duration	S	7/8/9	Full	1 test - fuel; 2 tests - oxidizer
	Tank Insulation		X	3	2 short/1 long	S	3/5/6	2.5/60/full	Test 3 objective is loading/long exposure. Test 5 is exposure and some vibration
	Stored Gas Mass/Operations	X		2	2 near full duration	S	6/7	Full	

Note: Space vehicle/altitude ignition vehicle requires modest additional testing for broader objectives.

NDH004.01

Table 15. Propulsion System Test Program Summary
Launch Vehicle - Ground Ignition

OBJECTIVE	TEST NUMBER								
	1	2	3	4	5	6	7	8	9
	TIME HOURS								
NON-FIRING									
Resonant Modal Vibration	4	4							
Engine/Tank Purge		1	1	1					
Propellant Loading Procedure		1	2	2					
Propellant Mass			3	3	3				
Engine/Feed System Chill		1	1	1					
Hardware Thermal			3a	3a	3a				
TOTAL	4	7	7	7	3				
FIRING									
Firing Duration, Seconds	n/a	n/a	2.5	15	60	Full	Full	Full	Full
Engine Thrust - % maximum	n/a	n/a	n/a	70	100	100	100 engine out	100 press valve out	100
FEATURES									
Component Environment Requirement					X	X	X		
Instrumentation Verification				X	X	X	X		
Hazardous Fluid Leakage					X	X	X		
Pressurization System Verification					b	b	b	b	
Propellant Margin Uncertainty						b		b	b
POGO Failure					b		b	b	b
Thrust Vector Control Failure					b	b		b	
Clustered Engine Performance			X	X		X			
Low Level Cutoff							X	X	X
Tank Insulation			X		X	X			
Stored Gas Mass/Operations						X	X		

NDH006.01

Note: a - Time not additive
b - Requirements affect test objective consolidation

For purposes of comparison, the MPTA program plan was for fifteen tests including three non-firing tests. Eight of the fifteen tests were to be for full duration; however, only six full duration tests were actually conducted.

The reduction in recommended numbers of tests from fifteen for MPTA to nine for current planning is realistic and is attributable to: (1) an engine which is more mature, of current technology, and less complex than was SSME; (2) availability of demonstrated analytical tools for facility water cooled deflector design; and (3) availability of improved analytical techniques for vehicle design in a number of areas, thus the need for less extensive testing. The number of test attempts to complete the planned program should also be less than for MPTA due to availability of a more mature and less complex engine thus fewer failures and application of "lessons learned" relative to control instrumentation improvements and fewer red line violations.

The test program in Table 15, as stated, may be considered the minimum program considering technical requirements, although the test program may not be compatible with minimum total program cost over the life of the program. This is because the minimum test program is developed to be compatible with the normally accepted method of developing/launching space vehicles—vehicle launch conducted and supported by the required, highly-trained technical personnel necessary to successfully accomplish the mission. This approach necessitates continual availability of significant numbers of highly-trained and specialized personnel which inevitably leads to significant program cost. An alternate approach whereby expanded propulsion system ground tests programs are conducted may lead to vehicle launch with smaller and less specialized launch crews. The acceptability of this approach has not been established nor is total program cost reduction assured. Repeated launch delays or the unnecessary loss of a single vehicle can affect total program cost and introduce extraneous factors which could best be avoided. Thus, careful evaluation is necessary should launch philosophy changes be contemplated.

PROPULSION SYSTEM TEST COSTS/BENEFIT DISCUSSION

The rationale for test and verification (T&V) is that reliability will be improved by detecting and correcting design or manufacturing errors before they can cause failure during operations. The 'merit' of a T&V program can be evaluated by comparing its costs (e.g., test facilities, test equipment, test articles, and test operations) against its benefits (e.g., economic value of reduced failures).

TEST AND VERIFICATION COSTS

T&V costs rough-order-of-magnitude (ROM) in new launch vehicle programs can be estimated from past program costs. According to Rockwell/STSD data, 'test' costs (not including government furnished facilities and equipment or other government costs) have averaged 4.9 percent of 'non-recurring' DDT&E costs across Gemini, Saturn S-II, Apollo CSM, and Space Shuttle Orbiter programs. Also, there have been only slight variations in 'test' costs as a percentage of 'non-recurring' costs, ranging from 4.2 percent for the S-II program to 5.2 percent for the orbiter program.

To the extent that 'test' cost data were recorded and are available at the subsystem level, they indicate that structural, flight control (including hydraulics), avionics and propulsion subsystems are characterized by higher than average test costs (as a fraction of subsystem DDT&E). For example, on the orbiter program, MPS test costs (excluding SSMEs) ran 8.3 percent of MPS DDT&E.

Based on historic T&V cost parameters, reasonable bounds on ROM T&V costs for a new (undefined) launch vehicle run from 4 to 6 percent of non-recurring DDT&E (e.g., a $5,000M DDT&E advanced launch system would include between $200M and $300M of T&V). Of that T&V cost, roughly 10 to 15 percent would likely be spent on MPS testing (e.g., between $20M and $45M would cover the MPS test effort, again exclusive of facilities, equipment and engines).

BENEFITS OF TEST AND VERIFICATION

The economic payoff of T&V efforts is lower 'loss'—the product of fewer failures during a system's operational phase or less serious consequences when failures occur. For example, if a $45M MPS T&V program for an unmanned, expendable advanced launch system were to prevent only one catastrophic failure during the fleet's life-cycle of 200 to 500 launches, it would have justified itself ten times over by avoiding the loss of a ~$50M to ~$100M launch vehicle, the loss of its ~$200M to ~$1,000M cargo, and the unavoidable waste of several hundred million dollars in 'fixed' costs during a period of fleet grounding.

The economic payoff of T&V in manned, reusable or 'civil' space programs is immensely higher than in unmanned, expendable programs since the consequences of a catastrophic failure are immensely more severe! For example, loss of Challenger (~$1,500M replacement value), its TDRS/IUS cargo (~$200M

replacement value), SRB re-design and 'fixed' costs of $3,000M - $5,000M during the stand-down, dwarf by a factor of twenty to thirty times the orbiter program's T&V outlays.

Perhaps other 'non-quantifiable', but real losses (the crew, the agency's prestige, and the subsequent shifts in national space launch policy) could have been avoided with more extensive T&V efforts earlier. One 'lesson learned' from the history of launch vehicle development is that, while failures are both inevitable and terribly expensive, early T&V efforts to detect and correct the sources of failure before they occur are both effective and relatively inexpensive.

PROPULSION COMMUNITY QUESTIONNAIRE

One of the major reasons for performing this study was to record the thoughts of many of the propulsion development engineers that have contributed so heavily to the success of Saturn and Shuttle—the "old hands" that are now contemplating new careers or retirement. A few Rockwell people have contributed heavily in this endeavor, but a questionnaire distributed throughout the NASA and NASA contractor propulsion community proved to be invaluable for preserving the judgement of managers and engineers with years of propulsion background.

The response to the questionnaire was gratifying. Almost all respondents indicated a high interest in the subject and took the time to fill out a rather lengthy questionnaire. Table 16 is the list of respondents. The actual questionnaire is reproduced as Appendix 11 and is included for the reader's information.

The responses to the question addressing the role of "all-up" system testing were almost consistent. These are included verbatim (except two which were paraphrased) in Table 17. In our view, the responses overwhelmingly make the case for propulsion system testing.

The author's professional views on this question are included separately on Pages 60 through 66.

ANALYTICAL HIERARCHY PROCESS

In addition to the primary questions, the questionnaire contained a number of opinion trade questions which were analyzed by an analytical hierarchy process (AHP). This process is an attempt to quantify the collective subjective opinions of the respondents. The objective is to establish priorities based on cost, schedule, and reliability from five system verification methods: systems analysis only; battleship testing; flight configuration article testing; flight-readiness firing at the launch pad; or flight test only. All of the verification methods assumed previous component and engine development testing.

The analytical method used to analyze the respondents' opinions is based on a book by Thomas L. Saaty, <u>Decision Making for Leaders—The Analytical Hierarchy Process for Decisions in a Complex World</u>, published in 1986. To quote the preface of this book, ". . . the AHP is a method of breaking down a complex, unstructured situation into its component parts; arranging these parts, or variables, into a hierarchic order; assigning numerical values to subjective judgements on the relative importance of each variable; and synthesizing the judgements to determine which variables have the highest priority and should be acted upon to influence the outcome of the situation." AHP is based on the fact that human reasoning works best when faced with only two choices. The process calculates preference weighted rankings of the different alternatives or choices available. (In this case, the alternatives are the various levels of testing.) Certain criteria are determined to be the most important factors in making a decision, and these are ranked according to relative importance among them compared to all other alternatives considering

strengths and weaknesses relating to each pre-determined criteria. After all judgements and rankings have been made, a matrix is computed showing an overall score for each choice. The program used for this process can also consolidate the opinions and judgements of many individuals into a group choice.

The results are shown in the accompanying table. The numbers in this table reflect the collective priorities based on the respondents' opinions. The absolute values of the numbers are not meaningful, but the comparison between the numbers for criteria and test methods is significant.

As was expected, reliability was the predominant criterion when compared to cost and schedule. Based on the reliability criterion, the highest priority test method was by flight configuration test article, followed by battleship testing. The priority assigned to the other choices was significantly lower, and the lowest ranking was given to analysis only.

Table of Results

Criteria: Which criterion is more important when planning a verification program for an advanced propulsion system?

Cost	Schedule	Reliability
.126	.102	.772

Test Method: Which test plan alternative will result in the most reliable propulsion system?

Analysis Only	Battleship Testing	Flight Configuration Test Article	FRF Testing	Flight Testing
.038	.211	.484	.123	.145

Table 16. Questionnaire Respondents

Adams, Alan / Pratt and whitney
Andrews, Byron / Rockwell
Baker, J. L. / Rockwell
Brasseaux, Hugh / JSC
Breedlove, Dave / Rockwell
Bruce, Jim / Rockwell
Burg, Roger / Rockwell
Burns, Robert / Rockwell
Carlson, Norm / KSC
Cobb, Bill / MSFC Retired
Coester, Steve / Rockwell
Connell, Don / Pratt & Whitney
Corn, Graydon / NASA Retired
Driscoll, Dan / NASA Retired
Driver, Orville / Calspan
Emerick, Bruce / Martin Michoud
Flauss, Emille / Martin Michoud
Fox, Ed / Martin Marietta Aerospace
(By George Elliott)
Glasgow, Lynn / Rockwell
Glasser, Sid / Rockwell
Harrison, Colin / MMC Denver
Harsh, George / MSFC
High, Dick / JSC
Hildebrand, Arnold / MSFC Retired
Hillbrath, Henry/The Boeing Co.
Hokanson, Roland / Martin Michoud
Jackson, Ed / Rockwell
Johnson, John / Martin Michoud
Johnstone, Harry / NASA Retired
Kaser, Ted / Rockwell
Kingsbury, James / The Boeing Company
Lamberth, Horace / LSOC
Lee, Sam / Rockwell
Lincoln, Bob / Rocketdyne
Lindsey, Bill / Rockwell
Lyddon, T. E. / Rockwell Retired
McCandless, Mark / Martin Michoud
McCool, Alex / MSFC
Mosbrook, Ed / Rockwell
Muller, Phil / MSFC
Norquist, Lawrence / Martin Marietta Corp.
Odom, Jim / NASA HDQS
O'Neal, Adrian / McDonnell Douglas
Owen, Jim / MSFC
Pearson, Jim / U.T.C
Platt, Gordon / MSFC
Pride, Bob / MSFC
Redus, Jerry / MSFC
Reid, Bill / Rockwell Retired
Ritter, Glenn / MSFC
Schaffer, Virgil / Rockwell
Schweikle, Dave / McDonnell Douglas
Smith, Carlyle / MSFC Retired
Smyly, Harold / MSFC
Stewart, Frank / MSFC Retired
Schwartz, Dick / Rocketdyne
(By Jerry Johnson)
Trott, William / Rockwell
Vizek, Bill / Pratt and Whitney
Winch, John / The Boeing Company
(By James Stansell)
Winstead, Tom / MSFC Retired
Witt, Don/ Pratt & Whitney
Wong, Dale / Rockwell
Wood, Charles / MSFC Retired
Wood, Jim / Eagle Engineering
Worlund, A. L. / MSFC
Zeise, Jim / Rockwell

Table 17. Tabulation of Questionaire Responses - Opinions of Role of All-Up System Testing

RESP. NO.	
92	"IF ANY ITEM IS GOING TO FAIL, HAVE IT FAIL ON THE GROUND FIRST WHERE IT CAN BE DIAGNOSED AND FIXED BEFORE FLIGHT"
14	"..SYSTEMS TEST PROGRAM PROVIDES SYSTEM VERIFIC-ATION AS CLOSE TO FLIGHT AS POSSIBLE WITHOUT JEOPARDIZING FLIGHT VEHICLE AND/OR CREW"
79	"VERY IMPORTANT", BUT THOROUGH COMPONENT TESTS REQUIRED BEFORE SYSTEM TESTING - I.E. SSME FASCOS (STILL NOT COMPLETELY FLIGHT QUALIFIED)
94	"ABSOLUTELY ESSENTIAL.."
97	"A NECESSARY COMPONENT OF A GOOD VERIFICATION PROGRAM"
7	"THE SYSTEM LEVEL TEST PROVIDES THE BEST OPPORTUNITY TO IDENTIFY AND FIX ERRORS, OVERSIGHTS, AND POOR JUDGEMENTS...IN DESIGN"
35	"..MANDATORY SYSTEMS LEVEL TEST PRIOR TO FIRST FLIGHT SHOULD BE A GIVEN"
105	"A FULL UP SYSTEMS TEST IS MANDATORY" - UNLESS NEW SYSTEM CAN BE QUALIFIED BY SIMILARITY TO EXISTING HARDWARE
125	"..AN "ALL-UP" SYSTEMS TEST THAT WILL TEST THE PROPULSION SYSTEM IN ALL ASPECTS TO "DESIGN MARGINS" IS ESSENTIAL."
29	"IT IS VERY IMPORTANT TO HAVE EMPIRICAL DATA FROM "ALL UP" SYSTEMS TESTING TO SUPPORT DESIGN VERIFICATION..."
71	"..SYSTEM LEVEL TEST CONFIGURATION PROVIDES AN OPPORTUNITY TO IDENTIFY HARDWARE AND SOFTWARE DESIGN CHANGES, MANUFACTURING ERROR CORRECTIONS, AND PROCEDURAL REFINEMENTS REQUIERD PRIOR TO FLT."
133	"ALL UP" SYSTEMS TESTING ON ANY FLIGHT ARTICLE IS MANDATORY FOR THE SUCCESS OF ANY MISSION"
19	"I CAN ONLY DESCRIBE IT AS A MANDATORY VALUABLE TOOL TO ANALYZE SYSTEMS PRIOR TO THE 1ST LAUNCH. I DON'T BELIEVE OUR PROGRAM COULD AFFORD TO LAUNCH A NEW SYSTEM WITHOUT PRIOR "ALL UP" SYSTEMS TESTING."
84	"GROUND TESTING IS THE KEY FOR FUTURE PROGRAMS, JUST AS IT WAS 40 YEARS AGO..."
54	"ALL-UP" SYSTEMS VERIFICATION TESTING MUST BE REQUIRED FOR ALL LARGE LIQUID PROPULSION SYSTEMS. THE VARIABLES AND UNKNOWNS ARE JUST TOO GREAT TO EXPECT ANY PRESENT ANALYSIS TECHNIQUES TO PREDICT THE INTERACTIONS ..."
48	"I FEEL IT'S MANDATORY TO HAVE ALL-UP SYSTEMS TESTING OF NEW PROPULSION SYSTEMS PRIOR TO 1ST LAUNCH, ESPECIALLY FOR MAN RATED SYSTEMS.
109	"MANDATORY"
22	""ALL-UP" SYSTEM TESTING IS EXTREMELY IMPORTANT TO ASSURE SAFETY AND RELIABILITY OF A NEW PROPULSION SYSTEM."
128	"AN "ALL-UP" SYSTEMS TEST IS HIGHLY DESIREABLE TO VERIFY OPERATIONAL PROCEDURES, PROPELLANT LOADING TECHNIQUES, SYSTEM OPERATIONAL LIMITS ..."
64	"I FEEL THAT "ALL-UP" SYSTEMS TESTING IS MADATORY"
46	"THIS TESTING IS NECESSARY TO BE CERTAIN THAT THERE IS NOT A TECHNICAL PROBLEM THAT HAS NOT BEEN CONSIDERED OR OVERLOOKED."
77	"ESSENTIAL- AND ALSO A COST AND TIME SAVER"

Table 17. Tabulation of Questionaire Responses - Opinions of Role of All-Up System Testing (Continued)

RESP. NO.	
90	"ABSOLUTELY REQUIRED. ONLY METHOD TO INSURE ALL COMPONENTS(PRESSURIZATION, PROPELLANT FEED, ELECT. AND ENGINE CONTROL)ARE WORKING PROPERLY TOGETHER."
100	"AN "ALL-UP" PROPULSION SYSTEMS TEST PROGRAM PROVIDES A MOST SIGNIFICANT INCREASE IN CONFIDENCE IN A NEW PROPULSION SYSTEM."
30	"ALL-UP TESTING IS NECESSARY WHEN- A)THE SYSTEM IS TOO LARGE FOR THE CONDITIONS AND ENVIRONMENT IMPOSED..." B)AND/OR: WHEN THERE IS A HIGH SENSITIVITY OF INTERACTION BETWEEN SUBSYSTEMS AND COMPONENTS."
26	"MANDATORY"
129	...OPERATIONAL ENVIRONMENT OF THE MANY COMPONENTS THAT MAKE UP A COMPLEX PROPULSION SYSTEM CANNOT BE ACCURATELY DETERMINED BY ANY MEANS OTHER THAN THE SYSTEMS TEST...
110	"EXTREMELY IMPORTANT FROM SAFETY/RELIBILITY ASPECTS- UNFORSEEN SYSTEM RESPONSES AND INTERACTIONS A HIGH PROBALITY-"
134	"I FEEL AN "ALL-UP" TEST IS NECESSARY. A LAUNCH PAD CAPABLE OF EXTENDED FRF TESTS WOULD SEEM TO MAKE A LOT OF SENSE."
135	"ALL-UP TESTING IS NECESSARY TO ESTABLISH COOL-DOWN METHOD, VERIFY START TRANSIENTS AND S/D SEQUENCE, VERIFY ENGINE SUPPLIED TANK PRESSURANT SYSTEMS, VERIFY PROPELLANT DEPLETION ..."
27	"SYSTEMS TESTS ARE NECESSARY TO VERIFY VEHICLE-ENGINE INTERACTIONS AND OPERABILITY."
23	SYSTEMS TEST REQUIRED TO VERIFY: RELATIVE IGNITION-C/O TIMING OF EACH ENGINE; COOLDOWN UNDER VEHICLE CONDITIONS; ACOUSTIC INTERACTIONS-

RESP. NO.	
56	"I BELIEVE AN "ALL-UP" SYSTEMS TEST IS MANDATORY FOR ANY NEW PROPULSION SYSTEM."
41	"AXIOM: THE MORE EFFORT (MONEY) PUT INTO THE INITIAL TEST PROGRAMS OF SYSTEMS AND SUBSYSTEMS, THE LESS COSTLY THE PROGRAM BECOMES-"
132	"MANDATORY OF MANNED VEHICLE- IF UNMANNED FIRST LAUNCH WITHOUT A SYSTEMS TEST MUST BE CONSIDERED A TEST FLIGHT WITH FAILURE AN ACCEPTABLE RESULT."
70	"ALL UP SYSTEM TEST IS ABSOLUTELY REQUIRED. SUB-SYSTEM TESTING DOES NOT ADEQUATELY VERIFY INDIVIDUAL SUBSYSTEMS INTERACTION."
108	"I CONSIDER "ALL-UP" SYSTEMS TEST TO BE MANDATORY TO VERIFY SYSTEMS DESIGN PARAMETERS. ENGINE SYSTEMS TESTING HAS SHOWN THAT INITIAL DESIGN CRITERIA BASED ON EXTRAPOLATED DATA MUST BE CONFIRMED BY EXTENSIVE ALL UP TEST."
106	"VERY IMPORTANT IF NEW TECHNOLOGIES OR SIGNIFICANT DEVIATIONS FROM PAST EXPERIENCE ARE INVOLVED."
85	"THE "ALL-UP" SYSTEM TESTING IS REQUIRED FOR NEW PROPULSION SYSTEMS BECAUSE IT SAVES LIVES AND FUNDING."
137	"INTEGRATED TESTING SHOULD BE WEIGHTED EQUALLY WITH LOWER LEVELS OF TESTING, SUCH AS ENGINE LEVEL AND COMPONENT LEVEL. INTEGRATED TESTING ALLOWS VERIFICATION OF INTERFACES WHICH DO NOT EXIST AT LOWER LEVELS."
60	"... ESSENTIAL TO VERIFY SPECIFIC VEHICLE SYSTEM.
136	"I BELIEVE "ALL-UP" SYSTEM TESTING TO VERIFY A NEW PROPULSION SYSTEM PRIOR TO FIRST LAUNCH IS MANDATORY."

Table 17. Tabulation of Questionaire Responses - Opinions of Role of All-Up System Testing (Continued)

RESP. NO.	
88	"I BELIEVE THE NEED FOR AN "ALL-UP" PROPULSION SYSTEM TEST IS DEPENDENT ON THE BASIS FOR DESIGN, HISTORY OF THE DESIGN, OR SIMILARITY WITH OTHER PROGRAMS STATE-OF-THE-ART, RISK, AND OTHER FACTORS. .... IF NEW TECHNOLOGY IS INVOLVED, SUCH AS A NEW ENGINE, NEW FORM OF PRESSURIZATION SYSTEM NEW COMPONENT DESIGNS, NEW FEED SYSTEM CONCEPTS, AN "ALL-UP" SYSTEMS TEST PRIOR TO FLIGHT IS MANDATORY. HOWEVER, I BELIEVE IN TESTING "FLIGHT" DESIGN RATHER THAN BATTLESHIP DESIGN."
15	"MANDATORY FOR A NEW SYSTEM. ALL ASPECTS MUST BE CONSIDERED, AND NOT JUST THE ENGINE, I.E. PROPELLANT LOAD, RECONDITIONING, OFF LOAD ... ETC"
130	"I CONSIDER THIS TYPE TESTING MANDATORY FOR NEW PROPULSION SYSTEMS(LIQUID & SOLID). MY EXPERIENCE ... HAS CONVINCED ME THAT ADEQUATE RESULTS OF SYSTEM PERFORMANCE CAN ONLY BE DEMONSTRATED FULLY WITH FLIGHT TYPE CONFIGURATION OF TEST ARTICLES."
2	REQUIRED TO:"-VERIFY HARDWARE RELIABILITY IN INTEGRATED SYSTEMS OPERATIONS;-VERIFY DESIGN INTENT VS. INTEGRATED OPERATIONS..;-CORRECT BY DESIGN REVISION AND RE-TEST ANY MALFUNCTIONS...;-DETERMINE COUNTDOWN TIMELINES;-DEVELOP SPECIAL PROCEDURES...;-REMOVE FACILITY TO GSE UNKNOWNS;-REMOVE INTEGRATED SYSTEMS PERFORMANCE UNKNOWNS"
112	"GOOD FOR VERIFICATION OF:A-PROPELLANT CHILLDOWN WITH FLIGHT SYSTEM;B-BOATAIL ENVIRONMENT EFFECTS..C-MULTIPLE ENGINE EFFECTS ON ACOUSTICS...ETC; D-TANK PRESSURIZATION SYSTEMS;E-PREVALVE SEQUENCING WITH ENGINE SHUTDOWN."
67	"POGO PULSING IN THE MPT PROGRAM WAS INSTRUMENTAL IN CLEARING THE POGO SYSTEM FOR THE STS-1 LAUNCH."
21	"THIS TYPE TESTING IS REQUIRED TO INSURE THAT THERE ARE NO UNPLEASANT SURPRISES WITH THE FIRST USEAGE OF A NEW SYSTEM...ALL-UP SYSTEMS TESTING IS MANDATORY TO PROVIDE THE REQUIRED CONFIDENCE LEVEL IN A NEW SYSTEM PRIOR TO ITS FIRST USEAGE.."
59	"I WOULD CONSIDER IT MANDATORY FOR 2 REASONS 1) THE TEST IS REQUIRED TO WRING OUT PROBLEMS o IMPACT OF NOT TESTING WOULD BE PROBABLE DOWNSTREAM LOSS OF A FLIGHT VEHICLE ..... 2) THE FIRST TIME THE ENTIRE SYSTEM IS PUT TOGETHER THERE ARE INEVITABLE SCHEDULE SLIPS FOR 'I FORGOTS' AND OTHER PROBLEMS. IF THE ALTERNATE IS FLIGHT TEST THIS MEANS MAJOR FIRST LAUNCH SLIPS WHICH ARE UNACCEPTABLE..."
74	""ALL-UP" TESTING VERIFIES COMPONENTS' COMPATIBILITY & LACK OF FAULTY INTERACTIONS."
96	"WERE I SCHEDULED TO RIDE ON A "NEW" LAUNCH VEHICLE, "ALL-UP" SYSTEMS TESTING WOULD BE A PRIMARY REQUIREMENT."
42	"AN ABSOLUTE NECESSITY.."
57	"ONE DANNOT AFFORD TO TAKE THE RISK OF LAUNCHING A VEHICLE ON THE BASIS OF ANANLYSIS AND FRF. THE LOSS OF ONE VEHICLE AND ASSOCIATED PAYLOAD WILL MORE THAN PAY FOR THE TEST FACILITIES AND THE TEST PROGRAM. THE BOTTOM LIN OF CHALLEGER WAS TO DO MORE TESTING."
137	"SYSTEM TESTING IS ESSENTIAL IN PROGRAM DEVELOPMENT TO: 1) ASSURE SYSTEMS OPERATION IN OPERATIONAL ENVIRONMENT. 2) REDUCE RISK ..IN SYSTEMS TRANSITION FROM DEVELOPMENT TO OPERATIONAL TO PRODUCTION. 3) THROUGH INTEGRATED TEST PLAN PROVIDE DATA AND EXPERIENCE FOR FLIGHT ENVIRONMENT ..... 4) PROVIDE SOFTWARE DEVELOPMENT" ..VERIFICATION

Table 17. Tabulation of Questionaire Responses - Opinions of Role of All-Up System Testing (Concluded)

RESPONDENT NO.	
138	"IN THE ALS PROGRAM THE ROLE OF "ALL-UP" SYSTEMS TESTING IS THAT OF AN ESTABLISHED APPROACH FROM PREVIOUS PROGRAMS WHICH, LIKE ALL SUCH APPROACHES, MUST BE EVALUATED FOR COST EFFECTIVENESS BEFORE ITS USE CAN BE JUSTIFIED. THE ROLE OF "ALL-UP" SYSTEMS TESTING IS STRONGLY DEPENDENT ON THE NATURE AND OBJECTIVES OF THE NEW PROPULSION SYSTEM. IN THE ALS PROGRAM THE GOAL IS COST REDUCTION BY "NEW WAYS OF DOING BUSINESS". (COST IN THE ALS PROGRAM IS DEFINED AS THE TOTAL PENALTY TO SOCIETY IMPOSED BY THE PROGRAM AND THEREFORE STRONGLY REFLECTS THE COSTS OF UNRELIAB-IILITY.) A MPTA IS REGARDED AS A "BUSINESS AS USUAL" APPROACH. SUCH APPROACHES ARE NOT ASSUMED IN THE ALS PROGRAM UNLESS SATISFACTORY SUPPORTING RATIONALE CAN BE DEVELOPED. THE PROPOSED ALS PROPULSION SYSTEM IS A PROGRESSIVE EVOLUTION OF PAST SYSTEMS. THE ALS SYSTEMS HAVE EXTREMELY HIGH RELIABILITY GOALS AND WILL THEREFORE BE DESIGNED WITH VERY LARGE MARGINS AND A GREAT DEGREE OF FAULT TOLERANCE. THE LIQUID ENGINES WILL BE OF MATURE DESIGNS THAT WILL BE EXTENSIVELY TESTED AT THE ENGINE LEVEL. MANY OF THE REASONS FOR "ALL-UP" TESTS IN THE PAST ARE NOT APPLICABLE TO THE ALS PROGRAM. -INADEQUATE FLIGHT INSTRUMENTATION - ADVANCED INSTRUMENTATION AND COMMUNICATIONS TECHNOLOGY WILL ENABLE ALL PARAMETERS WHICH COULD BE MEASURED ON THE GROUND TO BE MEASURED IN FLIGHT WITH EQUAL FIDELITY. -HARDWARE EXAMINATION AFTER TESTING - MANY ALS PROPULSION SYSTEMS ELEMENTS ARE DESIGNED TO BE RECOVERED AND REUSED, WHICH WILL ALSO ENABLE POST TEST EXAMINATION. -INADEQUATE SIMULATION OF DYNAMIC ENVIRONMENTS - GROUND FIRING OF ENGINES DOES NOT ADEQUATELY SIMULATE THE VEHICLE DYNAMIC ENVIRONMENTS, BUT IN PREVIOUS PROGRAMS, IN MANY CASES THIS WAS THE ONLY SOURCE OF ACOUSTIC OR VIBRATORY INPUT WHICH COULD BE APPLIED TO LARGE COMPONENTS AND ASSEMBLIES. SUITABLE SIMULATIONS ARE NOW AVAILABLE FOR ALL ELEMENTS OF INTEREST. -INTERACTION OF VEHICLE AND ENGINE SYSTEMS - ENGINE TESTS IN MANY PREVIOUS PROGRAMS DID NOT USE VEHICLE COMPONENTS DUE TO SCHEDULE CONSTRAINTS AND THESE INTERACTIONS WERE FOUND BY ALL UP SYSTEMS TESTS. IN THE ALS PROGRAM THESE COMPONENTS WILL BE INTEGRATED INTO THE ENGINE TESTING. -INADEQUATE COMPONENT TESTING/CONTROL OF PROCESSES - MANY SYSTEMS TEST FAILURES WERE THE RESULT OF QUALITY ESCAPES OR INADEQUATE COMPON-ENT TESTS WHICH WOULD HAVE BEEN FOUND EQUALLY AS WELL IN OTHER TESTING HAD IT BEEN SUFFICIENT. MANY TESTS ON FEW COMPONENTS VS. ADEQUATE TESTS ON A NUMBER OF COMPONENTS IS A KEY CONCEPT IN THE ALS PROGRAM. -LACK OF PREVIOUS EXPERIENCE WITH COMPARABLE SYSTEMS - MANY EARLY TEST PROGRAMS WERE JUSTIFIED BY THE LACK OF PREVIOUS EXPERIENCE ON COMPARABLE SYSTEMS. SUBSEQUENT ONES WERE BASED ON ESTABLISHED PRACTICE -THE ROLE OF "FIDELITY" IN SYSTEMS TESTING - IN THE PAST THERE HAS BEEN MUCH EMPHASIS ON "HIGH FIDELITY" OF TEST VEHICLES. ALL TOO OFTEN THIS HAS BEEN AS SUBSTITUTE FOR ADEQUATE ANALYSIS. TIME AND TIME AGAIN IT HAS BEEN SHOWN THAT FOR VARIOUS REASONS THAT RELIANCE ON TEST FIDELITY IS UNWARRANTED, ESPECIALLY WHEN EXTRAPOLATING FROM GROUND TEST TO FLIGHT TEST."

RESPONDENT NO.	
126	"NO DESIGN IS ADEQUATE UNTIL COMPLETE VERIFICATION IS ACCOMPLISHED. NOT JUST THE COMPONENTS, BUT THE MULTI-VARIANT RELATIONSHIPS BETWEEN HARDWARE & SUBSYSTEMS & PROCEDURES & FAULT TOLERANCE, ETC."
118	""ALL-UP" SYSTEM TESTING IS MANDATORY IF THE PROGRAM GOAL IS TO MINIMIZE PRE & POST-LAUNCH ANOMALIES/FAILURES USING SOUND RISK MANAGEMENT RATIONALE. THE AMOUNT OF SYSTEM VERIFICATION NEEDS TO BE TAILORED TO THE COMPLEXITY AND STATE OF TECHNOLOGY OF THE SYSTEM."
36	"ALL-UP SYSTEMS TESTING IS ESSENTIAL FOR VERIFIC-ATION OF A NEW PROPULSION SYSTEM PRIOR TO FIRST LAUNCH. UNEXPECTED PROBLEMS ALWAYS CROP UP AND REQUIRE CORRECTION PRIOR TO LAUNCH."
16	"IN THE PAST, BYPASSING ANY STEP IN THE INCREM-ENTAL DEVELOPMENT HAS COST MUCH MORE THAN TIME SAVED. A THOROUGH UNDERSTANDING OF EACH STEP & ITS RESULTS MUST BE GAINED BEFORE TAKING THE NEXT STEP."
131	"AN ALL-UP SYSTEM TEST PROVES THE DESIGNS AND PROCEDURES AND VALIDATES THE CRITERIA FOR LAUNCH SITES PRIOR TO USE WITH FLIGHT HARDWARE AND THE VERY VISIBLE PROGRAMMATIC ISSUES."
89	"MANDATORY! WE SHOULD DEVELOP ADEQUATE CFD CODES (COMPUTATIONAL FLUID DYNAMICS) TO "ANALYTICALLY DESIGN" THE ENGINE SYSTEMS BEFORE WE DEVELOP THE ENGINE COMPONENTS. THIS IS NOW IN THE RANGE OF TECHNOLOGY BUT WILL NOT REPLACE THE FULL UP ENGINE TESTING."
11	"MANDATORY. TESTING TO THE LIMITS OF ALL SYSTEM EXPECTED PARAMETERS SHOULD BE REQUIRED TO DEMON-STRATE LOADING CAPABILITY AND FLIGHT DURATION OPERATION."

THE ROLE OF PROPULSION SYSTEM TESTING — CHARLES C. WOOD

Is there a role for propulsion "system" testing for space vehicle development today in the high technology age? Managers of new space vehicle and military missile development programs are confronted with this issue. Considerable pressure confronts each program manager to reduce high, up-front costs of development programs. With limited ways to accomplish this low-cost objective, the omission of propulsion system testing may be one alternative to achieve desired funding reductions. Are such cost savings real or are they merely deferred to reappear later? Is program reliability impaired by such action or can it be achieved equally well by some other means?

More than 40 years devoted to the engineering aspects of space vehicle, military missile development and aircraft research in which some failures and many successes were achieved firmly imprints in my mind many important principals if success is to be achieved. My opinion regarding the desirability of propulsion "system" testing is unequivocally "yes," although circumstances may support carefully selected testing omissions. The following paragraphs discuss various aspects of propulsion "system" testing.

Propulsion "System" Test Requirements

Although exceptions are possible, propulsion "system" testing is essential for newly-designed manned vehicle programs; in the practical world, exceptions are limited. Circumstances which may permit propulsion "system" testing omission would be design similarity and use of state-of-the-art technology for all systems, both of which are unlikely for new vehicles. Prerequisites for vehicle programs without propulsion "system" testing are: (1) all systems must be thoroughly analyzed; (2) the interaction between systems must be thoroughly analyzed; and (3) the benefits of Flight Readiness Firings (FRFs) are thoroughly evaluated and implemented where shown to be beneficial. This approach is necessary because, as experience has shown, supposedly benign design changes frequently lead to serious operational or performance difficulties capable of inducing mission failure. Therefore, to avoid such unacceptable occurrences, propulsion "system" testing for manned missions is warranted when there is incorporation of a new engine system, a new pressurization system approach and hardware, or a new feed system design and hardware.

Objectives of a propulsion "system" testing program are: (1) to demonstrate that each separate system provides the required performance when interacting with other related vehicle systems; (2) to demonstrate that the structural capability of all systems is adequate; (3) to verify that methods devised for operations of all propulsion systems and the necessary supporting ground systems are correct, safe, and compatible; and (4) to show that ground systems have adequate capability and margin. Alternatives for determining flight worthiness of propulsion systems – analyses, FRF, and flight test programs – may be considered, and while they may provide program benefits, they also have major deficiencies. These alternatives are discussed in later paragraphs.

Aside from the engineering issues associated with propulsion "system" testing are other important issues – public and political images. A major concern, especially in manned programs, is public and thus political opposition. Launch site test programs highly visible to the public, which experience hardware failures, schedule delays and cancelled milestones are prime candidates for public distrust, unfavorable publicity and potential budgetary problems. The American perspective often overlooks the fact that failures are frequently necessary steps to achieve success. Propulsion "system" testing is less visible than FRFs, aborted or delayed flights, or in-flight failures with personnel aboard.

Propulsion "system" testing is not always essential for unmanned vehicle programs. From technical considerations, however, such testing is warranted, is highly beneficial, and is believed to be cost effective. Technical needs are the same as for manned vehicle programs. The public and political perceptions are similar but at a lower intensity, and the visibility of events can be purposely reduced. These features enhance the program manager's option to select alternative ways to develop and verify vehicle system performance to accomplish, if possible, front-end cost reductions and, potentially, total program cost savings.

System Verification by Analysis

Since initiation of the Apollo program, a technical revolution has occurred in the aerospace field.

- The United States industrial complex has become more heavily involved in space and missile programs.
- University graduates are better trained and have specialties in multiple aspects of aerospace science.
- Computers with almost limitless capability are available for increasingly complex analyses.
- Manufacturing machines and processes, as well as improved materials and methods of inspecting sophisticated products have all improved radically.

Ingredients thus exist today for providing vastly improved products. The Space Shuttle program, initiated 15 years after Apollo, capitalized on many of these improved capabilities, which are continuing to improve. Considering today's technological base, why can't system integration development and verification be accomplished by analysis? The answer to this question is provided in the following paragraphs.

Development and verification of a space vehicle/missile involves many disciplines and the interactive influence of each discipline. Examples of the various disciplines are: static and dynamic loads and their distribution, structural analysis and design, structural dynamics, acoustics, aerodynamics, fluid mechanics, fluid dynamics, mechanical design, thermal analysis and design, hydraulics, propulsion design, materials sciences (properties, corrosion, fracture

mechanics, crack propagation, life cycle, etc.), electrical power and distribution, power control, computers, and software. Many of the same disciplines are involved for supporting ground systems. This necessary and formidable array of talent is even more perplexing in that all disciplines must work together to assure success. Computers only perform as programmed; thus the real burden for success rests with design, analysis and system specialists. Assuming that the necessary knowledge exists to design each subsystem, does the knowledge exist to establish interactive influences of these many disciplines? Does management discipline exist within the system not only to permit but to mandate such interrelated activity? Do development schedules include sufficient time for such functions to be properly completed? Is the work environment of those who must participate buffered from every day pressures associated with normal development programs so that the in-depth, multi-disciplined thinking can be accomplished? Based on history from participation in more than 12 development and many research programs, the answer to most questions is **absolutely not!**

Oversight Examples

A few examples of "oversights" of consequence from past programs will emphasize this conclusion. Also included for improved insight are a few examples of data required from propulsion "systems" testing which are necessary for integration analyses. The examples are drawn from the Apollo, Skylab, and Space Shuttle experience.

1. On Skylab (the nation's only Space Station to date) one of the two cable trays running along the vehicle length was torn off the vehicle during powered flight. Included within the tray was one of the two stored solar arrays. In orbit, the Skylab's electric power was consequently less than planned, and the thermal balance within the Skylab was lost; the mission was salvaged only by almost super-human efforts. The provided cable tray vent area was either inadequate or the local aerodynamic pressures at the vent locations were greater than design values; thus vent flow was restricted and excessive pressures resulted.

2. During static firing, one S-IC stage experienced a hydraulic fluid-fed fire. Subsequent to successful test termination, two liquid oxygen (liquid oxygen) feedline geysers occurred and ruptured hardware, allowing 50,000 gallons of liquid oxygen to flow uncontrolled into the thrust structure and onto the engines. Failure of the test conductor to follow procedures and failure of the procedures to be properly emphasized relative to geyser prevention were responsible. Propulsion "system" testing benefits crew training and procedure development.

3. The need to revise design requirements for propellant position control within the liquid oxygen and liquid hydrogen tanks (which permits venting of gases during orbital operations) was detected late in the Apollo program, but fortunately early enough to save the program from failure and embarrassment. The technical issue involved was slight vehicle drag which repositioned propellants from the designed zero gravity propellant configuration, thus preventing gas-only venting. At that time, all related American research was based on zero gravity conditions for earth orbit operations. Consequently, no liquid vapor separator

existed to enhance workability of the designed system in the actual environment. After the facts were realized, the propellant control concept was changed, associated systems were redesigned with inclusion of much new hardware and deletion of old hardware, and a special flight test experiment vehicle was developed and flown to measure and photograph fluid behavior and establish system performance. The basic cause of this incident was inadequate perception of events that resulted in inadequate modeling, incomplete research throughout the country and the unfavorable influence of these factors upon the design specialist.

4. Pressurant diffuser structural failure in the Space Shuttle liquid hydrogen tank occurred several times. The diffuser was initially constructed of aluminum and component testing had been successfully completed prior to propulsion "system" testing. Structural failure of a diffuser in the liquid hydrogen tanks was unexpected (although not unusual in the liquid oxygen tank) as were similar failures for several modifications. The failure mechanism was flow induced vibration; a change in material from aluminum to steel was required to achieve a satisfactory design.

5. Structural testing of tankage on Space Shuttle detected deficiencies in both the liquid hydrogen and liquid oxygen tanks. Problem resolution was possible through major manufacturing changes which would be costly and result in significant schedule delays. Although contrary to the original vehicle design and operating philosophy, problem resolution was also possible by maintaining positive pressure within each tank commencing with propellant tank loading. Discovery of these problems coincided with initial propulsion "system" testing. Studies revealed vehicle vent valve controlled cycling (alternately opening and closing) during propellant loading and standby could be utilized to control tank pressure within specified ranges; costly tank changes and schedule delay were thus avoided. This approach for controlling tank pressure was developed and verified during propulsion "system" testing and the approach continues today. Unavailability of a propulsion "system" testing program would have delayed its development until flight hardware and launch complex completion permitted testing at the launch site. Program risk and cost would have been greater and significant schedule delay would have occurred.

Typical Data Requirements Examples

Three examples are included for which data are difficult to obtain other than from propulsion "system" testing.

1. Vehicle structural and propulsion system dynamic interaction (POGO) is a complex phenomenon which can have devastating consequences. Sophisticated analyses are necessary to establish stability margins and provide assurance of safe operation. These analyses are dependent, however, upon experimentally determined inputs which are generally the results of feed system data collected both with and without controlled input pulsing of the feed system during normal vehicle firing. These data are needed for various propellant levels within the tankage. This operation introduces risk and requires

modification to the vehicle flight configuration to accommodate the pulser and necessary special instrumentation.

2. This example represents a broad issue involving propellant management. Included are propellant loading, propellant volumes within the tanks and safe engine shutdown. Total data needs are unavailable from other sources unless full-duration FRFs are performed. Safe engine shutdown involves operation of the low level cutoff systems to assure adequate propellants are available during the shutdown cycle. Such testing involves risk to both vehicle and facility. Accurate propellant volumes are necessary for predicting vehicle performance. It is important because vehicle tankage volumes are calculated from drawing dimensions (including both plus and minus tolerances) while the finished tank volumes change with thermal exposure and loading. Propellant loading relies on accurate information to a known tank level. Involved are proper location of sensors, reliable hardware, understanding of propellant flow currents within the tanks, interaction of loading sensor electronics and ground computer systems, and software/hardware compatibility.

3. This example relates to preventing hydrogen gas venting during initial flight phases. At altitudes below approximately 50,000 feet, burning of vented hydrogen can result in unpredictable consequences. Involved are six independent factors: (1) aerodynamic heating generated by the vehicle configuration and trajectory; (2) vehicle insulation design which controls heat input to the hydrogen; (3) pressurization system performance and control system tolerance; (4) vent valve tolerance; (5) tank structural capability; and (6) liquid hydrogen stratification within the tank. Data on each of these are necessary to support analyses concerning hydrogen gas venting, and data on all but (1) are obtained from propulsion "system" testing.

<u>Other Considerations</u>

If propulsion "system" testing is not performed, management must be prepared to accept generous scheduling after flight vehicle arrival at the launch site preparatory to launch to accommodate problem discovery, fix determination, and probable retrofits. Furthermore, management must be prepared to deal with the critical press, political, and public environments resulting from inability to launch as planned and/or inflight anomalies. In spite of all design and integration analyses, component testing, and subsystem testing which have been conducted prior to arrival at the launch site, one should expect numerous malfunctions when initial ground and vehicle systems interface. This fact is even more pronounced when propellant flows begin as indicated in the preceding examples, which represent only a few of many possibilities which may have to be resolved before launch can occur. It may take weeks and several propellant loadings prior to complete resolution. Resolution of these many issues introduces the concern of ever-present safety factors which management must consider. Although cost has only been mentioned in passing, total program cost without propulsion"system" testing could easily approach or even exceed program cost had propulsion "system" testing been included initially.

System Verification by Flight Readiness Firing

This subject has been discussed indirectly to some extent in preceding paragraphs relating to analysis. It is now advantageous to consolidate thoughts.

As the name implies and as customarily conceived, the FRF assures vehicle readiness for flight and readiness of the launch complex for performance of required functions. The vehicle design, launch complex functions and integration aspects should normally have been proven elsewhere. The FRF for each new Space Shuttle vehicle, conceived on this basis, was fired for only 20 seconds, being more than adequate to accomplish its limited objectives.

For a new development vehicle involving new technologies and/or numerous dissimilarities from previous vehicles, an FRF can substitute for the propulsion "system" test program, although it can not be perceived as the limited test noted in the immediately preceding paragraph. In essence, the FRF must become a program itself—a propulsion "system" testing program conducted at the launch site. Such an approach will: (1) increase the time between program initiation and first launch due to delayed identification of problems and satisfactory resolution; (2) increase risk to the flight vehicle and the launch complex resulting from conduct of development tests, increased numbers of tests and longer duration tests; (3) result in higher program cost because of greater launch facility complexity, increased run requirements and imposed additional safety considerations; (4) measurably distract launch site personnel from accomplishing normal launch site functions; and (5) most importantly, reduce reliability of the vehicle and endanger the flight crew. This approach is not recommended generally, although for vehicle programs having considerable similarity with previous programs, it may be an acceptable and effective approach.

The FRF or "mini" propulsion "system" testing program will necessarily involve one to six propellant loadings and hot firing tests or more. The goal of the expanded FRF program continues to be verification that the vehicle is ready for launch. The issues discussed under the analysis section (and many others which were not included) portray the real world—development activities must precede verification for launch.

System Verification by Flight Testing

The final issue to be considered is flight vehicle test programs—development programs without the customary propulsion "system" testing and including the normal FRF. The FRF would resolve the many preflight development type issues and those related to short duration firings but would not include long duration firing. This approach is unacceptable for manned flight programs. The approach may be acceptable for unmanned programs. For unmanned programs, prerequisites for technical acceptance are thorough analysis of all systems individually and then as integrated systems. Management prerequisites are also involved and include all issues relative to the FRF in the absence of the normal propulsion "system" testing program. These include cost, risk, public image, and other issues. The subject

approach and postulated loss of an unmanned vehicle could easily result in total program cost greater than initial inclusion of a formal propulsion "system" testing program.

CONCLUSIONS AND RECOMMENDATIONS

Propulsion system development test programs for the seven Saturn stages, Space Shuttle, and some Air Force programs have been reviewed and evaluated for the purpose of documenting significant results of those various programs for future reference. Program benefits attributable to propulsion system testing are emphasized. Key technical design parameters are identified and program risks associated with curtailing propulsion system testing for each design parameter is discussed. The following conclusions and recommendations result from this study.

CONCLUSIONS

1. Propulsion system testing has prevented several catastrophic flight failures.

2. The complexity of interactive characteristics of the propulsion, structural and electrical systems defies accurate analytical representation. Propulsion system testing provides the necessary test data for "model basing" thus enhancing system analysis techniques.

3. A propulsion specialist survey characterizes propulsion system testing to be "mandatory."

4. Propulsion system testing determines hardware integrity and functional performance in the best possible environment excluding flight. Testing also certifies the environments utilized for component development and qualification.

5. Propulsion system testing integrates vehicle and ground hardware and procedures for propellant loading, safing, and firing operations for all systems.

6. Propulsion system testing provides a resource for determining stage/engine design margins, developing procedures and timelines and confirmation of extrapolated criteria used in engine development.

7. Potential risks for catastrophic flight failure, mission loss failure, vehicle hardware damage, and launch complex damage is reduced by propulsion system testing.

8. Potential risks for a delayed initial launch and subsequent launches is significantly reduced by propulsion system testing.

9. Propulsion system testing is required for new stage designs, advanced technology concepts introduced within major systems of existing designs, and possibly for existing designs modified to accommodate one or more major system redesigns. Full duration test firings are normally needed to satisfy system test requirements.

10. The "economic payoff" of propulsion system testing in manned and reusable space programs is higher than in unmanned, expendable programs since the consequences of catastrophic failures are more severe.

11. Propulsion system testing which prevents loss of only one vehicle may be considered cost effective to the program. For some programs, cost effectivity may be realized without vehicle loss. For unmanned, expendable stages, a program cost saving ratio for a single failure event may vary from near one to five or greater dependent upon cost of lost payload, fleet grounding, and other factors.

12. Factors to be included in propulsion system testing decisions are loss of life potential, cost, schedule, state-of-the-art design and procedures, history of design, design similarity with other programs, test site capabilities, and the agency's prestige and subsequent shifts in the national space policy in the event failure results.

13. Various methods for satisfying necessary program test and verification requirements are a separate development test vehicle, the normal or modified flight readiness program using flight hardware and a flight test program. Each development program must establish necessary test requirements and seek the single or combination of approaches which best satisfy established requirements.

RECOMMENDATIONS

1. New stages involving designs dissimilar to previous stage designs, stage redesigns incorporating advanced concepts in major systems and/or including advanced technology designs, hardware and/or materials shall include propulsion system testing. Program managers shall assess and establish programs for satisfying requirements including crew/passenger safety.

2. New stages similar to existing designs which exclude new technology may require propulsion system development and verification testing. Program managers shall assess and establish programs for satisfying requirements.

3. Propulsion system test hardware and test facilities necessary for conducting propulsion system testing shall be retained for rapid activation to firing status commensurate with resolving failures/anomalies from the early flight program.

4. Vehicle propulsion system test hardware and facility interfacing hardware shall duplicate flight vehicle/launch complex hardware wherever possible. Engineering's concurrence is suggested for usage of substitute hardware. Launch site operating procedures shall also be duplicated wherever possible.

5. Test hardware maturity standards shall be established as a prerequisite for propulsion system testing initiation. This may enhance safety, reduce risks to hardware and test personnel, and reduce cost.

6. Accurate and reliable instrumentation is vital for any successful test activity. Operating margin for hardware and systems may be the basis for reliability enhancement. Testing within "actual" environments for which hardware must perform is crucial to success. These three elements shall be evaluated and incorporated realistically in test programs to the maximum extent practical.

7. Development of a document which would serve as a guideline for all testing—qualification, verification, and acceptance test of components and systems—to demonstrate margins for workmanship and performance purposes would be beneficial and should be considered.

APPENDIX 1

REFERENCES

A. SATURN GENERAL

"Stages to SATURN" a technological history of the APOLLO/SATURN Launch Vehicles by Roger E. Bilstein.

AIAA paper No. 67-234, "Ground Testing of SATURN Launch Vehicle Stages." By Dan Driscoll. (see Appendix 10)

SATURN Illustrated Chronology, "SATURN's First Ten Years, April 1957 through April 1967."
MSFC document "SATURN IB Liquid Hydrogen Orbital Experiment."
Document No. R-ASTR-S-1-66, "AS-201 Technical Information Summary."
R-P&VE-PMP Document "Propulsion Systems Handbook."

B. S-I/IB DEVELOPMENT TEST REPORTS

Preliminary Summary Data Report - SAT-14
Preliminary Summary Data Report - SAT 15
Preliminary Summary Data Report - SAT-16
Preliminary Summary Data Report - SAT-17 and SAT-18
Preliminary Summary Data Report - SAT-19
Preliminary Summary Data Report - SAT-20
Preliminary Summary Data Report - SAT-21
Preliminary Summary Data Report - SAT-22
Preliminary Summary Data Report - SAT-23
Preliminary Summary Data Report - SAT-24
Preliminary Summary Data Report - SAT-25
Preliminary Summary Data Report - SAT-26-27-28, Volume 1
Preliminary Summary Data Report - SAT-26-27-28, Volume 2
Preliminary Summary Data Report - SAT-29
Preliminary Summary Data Report - SAT-30
Preliminary Summary Data Report - SAT-31

C. S-I ACCEPTANCE TEST REPORTS

Preliminary Summary Data Report - SA-01
Preliminary Summary Data Report - SA-02
Preliminary Summary Data Report - SA-03
Preliminary Summary Data Report - SA-04
Acceptance Firing Tests of Stage S-I-3
Static Test of Vehicle S-I-2
Preliminary Summary Data Report - SA-05
Acceptance Firing of Saturn Flight Stage S-I-3
Preliminary Summary Data Report - SA-06

Preliminary Summary Data Report - SA-07
Acceptance Firing of Saturn Flight Stage S-I-4
Preliminary Summary Data Report - SA-08
Preliminary Summary Data Report - SA-09
Preliminary Summary Data Report - SA-10
Acceptance Firing of Saturn Flight Stage S-I-5
Stage S-I-5 Acceptance Firing Tests
Preliminary Summary Data Report - SA-11
Preliminary Summary Data Report - SA-12
Preliminary Summary Data Report - SA-13
Acceptance Firing of Saturn Flight Stage S-I-6
Preliminary Summary Data Report - SA-14
Preliminary Summary Data Report - SA-15
Acceptance Firing of Saturn Flight Stage S-I-7
Preliminary Summary Data Report - SA-16
Preliminary Summary Data Report - SA-17

D. S-IB ACCEPTANCE TEST REPORTS

Saturn S-IB-1 Stage Final Static Test Report
Saturn S-IB-2 Stage Final Static Test Report
Saturn S-IB-3 Stage Final Static Test Report
Saturn S-IB-4 Stage Final Static Test Report
Saturn S-IB-5 Stage Final Static Test Report
Saturn S-IB-6 Stage Final Static Test Report
Saturn S-IB-7 Stage Final Static Test Report
Saturn S-IB-8 Stage Final Static Test Report
Saturn S-IB-9 Stage Final Static Test Report
Saturn S-IB-10 Stage Final Static Test Report
Saturn S-IB-11 Stage Final Static Test Report
Saturn S-IB-12 Stage Final Static Test Report

E. S-IC DEVELOPMENT TEST REPORTS

Summary Data Report S-IC-01-02-03
Summary Data Report S-IC-04
Summary Data Report S-IC-05
Summary Data Report S-IC-06
Summary Data Report S-IC-07, S-IC-08
Summary Data Report S-IC-09, S-IC-10
Summary Data Report S-IC-11
Summary Data Report S-IC-12
Summary Data Report S-IC-13
Summary Data Report S-IC-14
Summary Data Report S-IC-15
Summary Data Report S-IC-20, S-IC-21 and S-IC-22
Boeing Document No. D5-13509, Vol. V, S-IC-T Summary Test Report
Boeing Document No. T5-6777 S-IC-T/4 Static Firing Report

F. S-IC ACCEPTANCE TEST REPORTS

Saturn S-IC-1 Stage Final Static Test Report
Saturn S-IC-2 Stage Final Static Test Report
Summary Data Report S-IC-16, S-IC-17 (S-IC-1 Acceptance Tests)
Summary Data Report S-IC-18 (S-IC-2 Acceptance Test)

G. S-II DEVELOPMENT TEST REPORTS

Saturn S-II Summary Data, North American Aviation, Inc.
SID 65-39 Saturn S-II Static Firing Test Report No. 1
SID 65-814 Static Firing Test Report No. 2
SID 65-1215 Static Firing Test Report No.3
SID 66-374 Static Firing Test Report No. 4
SID 67-684 Extended S-Ii Battleship Program Final Test Report
S-II Battleship Weekly Test Reports, December 1965 - March 1966
S-II Battleship Monthly Test Reports, March 1966 - May 1967
SID 66-967 Saturn All Systems (S-II-T) Stage Static Firing Test Report

H. S-IVB REPORT

Douglas Report SM-46973

I. TITAN REPORTS

Titan II Missile N-1 Flight Test Report
The Gemini Program Launch Systems Final Report No. TOR-1001 (2126-80)-3

J. SHUTTLE MAIN PROPULSION TEST REPORTS

Rockwell International briefing charts entitled, "Main Propulsion Test Program, NSTL."
History of MPT Test Program, dated February 9, 1986
MPT Tanking Test History, dated July 1983
Quick Look Test Report - MPTA Static Firing No. 1 - Tests MPT-S1-001 and MPT-S1-002
Quick Look Test Report - MPTA Static Firing No. 2 - Test MPT-S2-001
Quick Look Test Report - MPTA Static Firing No. 3 - Test MPT-S3-001
Quick Look Test Report - MPTA Static Firing No. 4 - Test MPT-S4-001
Quick Look Test Report - MPTA Static Firing No. 5A and 5 - Tests MPT-5A and MPT-5
Quick Look Test Report - MPTA Static Firing - Test MPT 6-001
Quick Look Test Report - MPTA Static Firing Tests - MPT 6-02 and MPT 6-03
Quick Look Test Report - MPTA Static Firing Tests - MPT SF7-001
Quick Look Test Report - MPTA Static Firing Tests - MPT SF7-02
Quick Look Test Report - MPTA Static Firing Tests - MPT SF8
Quick Look Test Report - MPTA Static Firing Tests - MPT 9-01 and 9-02
Quick Look Test Report - MPTA Static Firing Tests - MPT 10-01
Quick Look Test Report - MPTA Static Firing Tests - MPT 11-01

Quick Look Test Report - MPTA Static Firing Tests - MPT 11-02
Quick Look Test Report - MPTA Static Firing Tests - MPT 12

APPENDIX 2

RATIONALE SUPPORTING NON-TEST RISKS ASSESSMENT

An overview of the material presented in this appendix has been presented in the main body of this report. Tables 2-1 and 2-2 are repeated here for reader convenience and summarize the results in tabular form (Tables 11 and 12 in main body). This appendix provides additional insight relative to the risks assignments for each subsystem/discipline evaluated. The risks relate to propulsion system development without propulsion system testing. The judgement classes for risk are extremely high, high, moderate, low, and minor and are based on frequency of occurrence in previous propulsion systems test programs, evaluation of analysis and design capabilities, characteristics of stage design, including redundancy, operational method, and other considerations.

"Wrong" Component Verification Requirement. The risk of early catastrophic inflight failures due to components being verified to the "wrong" requirements is judged to be extremely high. Examples of previous test failures of this type are the hydrogen pressurization system diffuser failure in the Shuttle hydrogen tank (potential hydrogen tank structural collapse due to insufficient ullage pressure) and the SSME main fuel valve failure in the orbiter aft compartment (compartment structural failure and fire damage).

Component operating and verification requirements are established well before the propulsion system is assembled and operated. Computer simulation analyses are well developed and are used to establish loads, pressures, operating temperatures and flow rates. Vibroacoustic environments induced by the various subsystems or components are difficult to manage, as are interactive influences of various environments and subsystems. Validity of requirements depends on the knowledge and experience of the design group, and yet the best of design groups are unprepared for these hundreds of components in such complex environments.

Correct requirements are not known completely until a system test program is conducted, including efforts required for data analyses and comparisons to individual component requirements. It can be stated that no component is verified until it has been subjected to a systems test program. FRF type firings can help to quantify environments for components, although availability of such data is late, and it must be carefully evaluated which results in delays. The flight instrumentation program must be significantly enlarged and the complex/costly launch site is exposed to additional risks.

Instrumentation Performance. The risk of launch delay is judged to be extremely high, the loss of mission and other considerations moderate. Launch scrubs and resultant schedule slips will result from inability to verify launch constraints due to failed (actual failure plus inaccuracies due to environment) instrumentation. Inability to analyze anomalies and inflight failures due to high instrumentation failure rates may be expected, and decision-making capabilities necessary to prevent failures will be adversely affected.

Table 2-1. Propulsion System Program Assessment - No Propulsion System Test Vehicle Shuttle Type Tankage

							PLANNED				POTENTIAL		
EVALUATION CRITERIA	VEHICLE FLIGHT CATASTROPHE RISK	MISSION LOSS RISK	FLIGHT HARDWARE DAMAGE RISK	LAUNCH COMPLEX RISK	LAUNCH DELAY RISK	MPTA TEST EXPERIENCE	DELTA COMPONENT PROGRAM IMPACT	DELTA SUBSYSTEM PROGRAM IMPACT	DELTA FLIGHT INCREASES BEFORE OPERATIONAL	DELTA FLIGHT INSTRUMEN-TATION INCREASE	DELTA FLGHT INCREASES BEFORE OPERATIONAL	20 SECOND FRF PROBLEM RESOLUTION REMAINING RISK	REMARKS
"Wrong" Component Verification Required	Very High	Very High	Yes	High	High	Yes	No	No	No	Yes	High	Low	Delta instrumentation for enviornment determined
Instrumentation Failure	Moderate	Moderate	Yes	Moderate	Very High	Yes	No	No	No	No	Low	Minor	FRF time inadequate for all measurements
Hazardous Fluid Leakage	High	High	Yes	High	Very High	Yes	No	No	No	No	Low	Moderate	FRF time provides inadequate exposure
POGO Failure	Moderate	High	Yes	Minor	Minor	No	No	Yes	No	Yes	Minor	Moderate	
Thrust Vector Control Failure	Low	Low	Yes	Minor	Low	No	No	No	No	Yes	Minor	Minor	
Pressurization System Performance	Moderate	High	Yes	Minor	Minor	Yes	No	Yes	No	Yes	Moderate	Moderate	FRF time inadequate
Propellant Mass Uncertainty	Minor	Moderate	Yes	Minor	Very High	Yes, very high	No	Yes, if possible	No	Yes	High	Low	
Propellant Loading Procedures/Operations	No	No	High	High	Very High	Yes, high	No	Yes	No	Yes	No	No benefit	
Clustered Engine Performance	Minor	Minor	Yes	Minor	Minor	No	No	No	No	No	Minor	Minor	
Low Level Cutoff Sensor	Minor	Minor	Yes	No	Moderate	Yes, high	No	No	Yes (unmanned) No (manned)	Yes (unmanned) No (manned)	Yes (unmanned) No (manned)	No benefit	Inflight test on manned flights not feasible
Performance Margin Uncertainty	Minor	High	No	Minor	No	No	No	No	Yes (unmanned) No (manned)	Yes (unmanned) No (manned)	Yes (unmanned) No (manned)	Moderate	Inflight test on manned flights not feasible
Engine/Feed System Chill	Minor	Minor	Yes	Minor	High	Yes, high	No	No	No	Yes	No	Minor	Loading test benefits
Tank Insulation	Minor	Minor	No	Minor	High	Yes	No	No	No	Yes (ground)	No	Minor	Loading test benefits
Hardwre Thermal Control	Minor	Minor	No	Minor	High	Yes	No	No	No	Yes	No	No benefit	Loading test benefits
Stored Gas Mass, Loading, Operations	Minor	Minor	Yes	Moderate	Minor	No	No	No	No	No	Minor	Minor	

Table 2-2 Propulsion System Program Risk Assessment — No Propulsion System Test Vehicle — Non-Shuttle Type Tankage — Risk Relative to Shuttle

EVALUATION CRITERIA	SHUTTLE FLIGHT CATASTROPHIC AND LAUNCH DELAY RISKS	DIFFERENCE FROM SHUTTLE					
		LARGER - SIMILAR L/D TANKS	SIMILAR VOLUME BUT COMMON BULKHEAD	SMALLER VOLUME BUT SIMILAR - ALTITUDE START	SMALLER VOLUME, COMMON BULKHEAD, ALTITUDE START	SMALLER VOLUME, COMMON BULKHEAD, ORBITAL START	SHUTTLE VOLUME, SIMILAR L/D, DIFFERENT PROPELLANTS
Pressurization System Performance	Moderate/ Minor		Higher/	Higher/	Much Higher/	Significantly Higher/Higher	*Higher/*Higher
Propellant Mass Uncertainty	Minor/ Extremely High	⇑ SAME	Higher/	/	Higher/	Much Higher/	*Higher/*
Low Level Cutoff	Minor/ Moderate	AS	/	/	/	/	*Higher/*
Engine/Feed System Chill	Minor/High	SHUTTLE	/	Higher/	Higher/	Significantly Higher/Higher	*Higher/*
Tank Insulation	Minor/High	⇓	Higher/	/	Higher/	Much Higher/	*Higher/*
Hardware Thermal Control	Minor/High		/	Higher/	Higher/	Significantly Higher/Higher	*Higher/*

NOTE: 1. Blank spaces denote same as Shuttle—column 2.
2. *Varies with propellant selection.

The time required to correct instrumentation deficiencies and to verify operational acceptability is usually extensive. All early propulsion system operations experienced multiple instrumentation failures. For MPTA, instrumentation failures resulted in cancellation of two planned firings. Terminal count interruptions/delays necessary to devise new procedures, to work around instrumentation failures, and to continue terminal count were also necessary on numerous occasions. Similar problems occurring at the launch site would have resulted in launch cancellations. The judged mission loss risk would be extremely high, except that most critical sensors are normally redundant.

Ground test allows the use of drag on instrumentation with wide band width capability. Flight vehicle instrumentation, on the other hand, is classically low band width to verify normal expected operation of systems only. This is a fall out of the telemetry total band width which precludes high frequency monitoring of a large number of measurements. The ground test operation allows "quick response" with additional measurements as needed as opposed to the flight systems which have a long turnaround time.

Propellant, Hydraulic Fluid and Hot Gas Leaks. The risk of an early catastrophic and mission loss due to an inflight leak are judged to be high. The risk to the launch complex is high, and the potential for launch delay is extremely high. The quality of design depends on the knowledge and experience of the design group.

Component and subsystem testing cannot simulate all thermal, vibration, and transient loads of the total system. Fatigue failures, fracture mechanics, seal compression, material creep aspects of design, and various manufacturing issues are better understood today than previously, but various failure modes still plague even mature systems. All programs using cryogenic propellants have experienced serious leaks or component/line failures in early testing. Vehicle designs have fluid connections ranging in size from one-quarter inch diameter to approximately eighteen inches in diameter. The Saturn V launch vehicle had between four and five thousand connections, while the S-IVB stage, which had only one engine, had between four and five hundred connections.

Propulsion system testing accomplishes many necessary functions such as resolving problems with the hazardous gas detection system used to monitor for safety; determining the overall leak characteristics of the vehicle through repeated exposure during loading and long duration firings, and identifying consistent "problem leakers" permitting special action such as changing seal material, increasing sealing load, and changing line supports.

Short duration FRF type testing may reduce risks of vehicle and mission loss moderately. It does not influence launch delay risk.

POGO Effect. The risk of an early catastrophic inflight failure due to POGO is judged to be moderate and loss of mission risk is high. Launch delay risk is judged to be moderate. Natural longitudinal deflection frequencies of the vehicle structure are predictable with current computer simulation programs and test data obtained

from independent structural test articles. Engine pump gains and thrust oscillations due to inlet pressure fluctuations can be determined by single engine testing which incorporates provisions for controlled pressure inputs/disturbances within the feedline upstream from the engine inlet. Complex computer simulations using noted inputs can be used to predict vehicle stability margins, although the accuracy of such predictions, and thus the acceptability, from a program standpoint is debatable.

Testing of flight type hardware with feed system pulsing is a likely requirement. Structural system and propulsion system frequency reinforcement in the lower modes will likely require hardware inclusion to ensure safe vehicle operation. Functional aspects of such hardware systems also must be demonstrated prior to flight. Such developments would add significantly to launch delay risk and to vehicle and launch complex risk. Such testing should be conducted remote from the launch site.

Thrust Vector Control (TVC) Performances. The risks of early catastrophic inflight and mission loss failures due to loss of TVC are judged to be low. This judgement is based on the assumption that TVC and avionic subsystem testing will be performed, that on-vehicle integrated testing and checkout will be performed without the engines firing, and that single engine testing with live TVC actuators will be performed. The remaining risk judgement is based on the fact that analysis would have to be used to verify that matching frequencies did not exist between the structure and TVC system and to verify that structural deflections did not allow nozzle collisions during startup and at high gimbal rates. The S-I stage structure had to be redesigned to decouple structure/TVC matching frequencies.

Pressurization System Performance. The risks of early catastrophic and mission loss failures due to pressurization system performance inadequacies are judged to be moderate and high respectively. This subject involves many disciplines requiring complex analyses such as feed system pressure loss, propellant heating during ground and flight including propellant stratification characteristics, the storage and heat sources for pressurant gasses, and time dependent pressurant requirements.

The system also requires complex mechanical and electrical hardware which must function near flawlessly under severe thermal, pressure, and vibration environments. Failure of the system to function as designed may result in inadequate NPSP to engines (explosion potential), inadequate tank pressure for tank structural capability (tank structural failure) and/or excessive tank pressure requiring flight venting of propellant tanks (potential vehicle fire damage) if vented below critical altitude.

Overall system complexity, considering the number of complex subsystems/components which must function as a unit and the physical number, scope, and complexity of analyses involved in sizing all the systems, is the basis for quantifying the risks. Independent component/subsystem development testing is expected to be part of any development program but is judged inadequate because of the limited simulation of such a complex system. Previous vehicle

designs clearly show this to be the most troublesome vehicle system. Supporting material is provided in several of the appendices

Vehicle simulation computer analyses of required pressurant flow rate versus time, feed system pressure loss, heat inputs to propellants and resultant stratification, and other type analyses for Shuttle are mature (after an extensive ground test program and many flights) and are possibly adequate for new programs using similar tank configurations. However, these analytical approaches are not mature for significant departures in tank configuration, propellant type, and other potential variables. Missions/designs requiring multi engine start or start in earth orbit are additional complicating features and affect risk.

Short duration FRF type testing provides some very valuable information and may reduce risk to some degree, however, needed answers to all requirements are not provided and the remaining risk is judged to be moderate.

Propellant Mass Uncertainty. The risk of early catastrophic and mission loss failures due to launch with inadequate propellant mass aboard are considered minor and moderate respectively. Launch delay risk is judged to be extremely high and unaffected by an FRF type firing and the companion propellant loading test conducted late in the program at the launch site.

Risk, as for pressurization, is based upon overall complexity of the operation and not on any single, technical issue. Involved are properly located sensors within propellant tanks, appropriate flow shields to avoid disturbing flow currents capable of adversely affecting sensor output, propellant temperature for determining density and controlling computer electronics, and calibration data to permit integration of vehicle sensors and the ground based control system. Propellant loading to acceptable accuracies for Space Shuttle, typical of any vehicle loading requirement, necessitated extensive testing on the MPTA. Data was collected on six different occasions with propellants aboard for a total time of thirty hours. Additional data were obtained for each vehicle loading. Vehicle instrumentation, in addition to flight instrumentation, is required to collect necessary data.

Risk is relatively unaffected by vehicle configuration or propellants, although vehicles requiring orbital restart may have higher risk.

On-board propellant mass at liftoff can be verified indirectly by data from initial flights if a performance penalty on these flights is acceptable to make ample allowance for loading inaccuracies, although such an approach may delay achieving vehicle operational readiness. Under these conditions the risk is judged to be minor, although launch delay risk remains high.

Propellant Loading Procedures/Operations. The risk of damage to flight hardware and facility hardware due to loading procedural complexity, errors and hardware oversights is judged to be high. The risk of unacceptable launch delays, while these procedures are being refined and hardware changes are made, is judged to be extremely high. Even with the Saturn and Space Shuttle propulsion

system test programs, loading problems plagued the first few Shuttle launches. The resultant delays would have been worse without the system test data base.

The high risk assessment is based on the fact that examples of damage to test hardware abound. The "close-call" the first time the external oxygen tank was drained at MPT due to excessive flow rates and pressure surges is an example. Basic tank structural or other deficiencies are frequent occurrences necessitating changes in some phase of vehicle operation to avoid cost increases and schedule delays. Such changes generally increase complexity and risk. Shuttle was no exception—tanks were loaded under pressure which was contrary to original design philosophy. Also, early in the program initial propellant quantities loaded aboard the vehicle were controlled by newly developed lower tank bulkhead thermal criteria. On the Saturn V S-II stage and earlier stages, hardware damage at the launch site included destruction of liquid oxygen tank vortex devices and feedline inlet screens. These failures were extremely hazardous—exposing raw, torn aluminum to both liquid and gaseous oxygen.

Clustered Engine Performance and Low Level Cut Off Sensors. Both evaluation criteria are judged to be minor in all risk categories except low level sensors for which launch delay risk is judged to be moderate. An FRF of short duration benefits only clustered engine performance as propellant levels prevent low level cutoff sensor activation. Increased numbers of flights, prior to achieving operational status, are likely for unmanned programs while the total operational concept is not feasible for manned flight programs.

Clustered engine performance uncertainty can be verified in flight with minor risks, although acceptance of a performance penalty may be required. Concerns are feed system interactions and degradation of performance.

Minor flight catastrophic and mission loss risks for low level cutoff sensors is because sensors are backup cutoff devices and other anomalies are required for these sensors to be needed. Later flight program anomalies may be expected to require cutoff sensors as payload demand decreases performance margins. No safe inflight data gathering effort for these sensors is known. Acceptance of identified risk assumes ample propellant allowance can be made on early flights and that flights are unmanned.

Large vehicles usually have large diameter, long feedlines which contain large masses of propellant residuals if propellants are not burned until levels are well below the feedline inlet. Propellant levels within such feedlines vary rapidly with time, thus the controlling system must be fast response and dependable. For MPTA, extensive difficulties were experienced initially with the system and serious flight failures would have resulted had its use been required and had development testing not been performed. Technical issues involved sensor inadequacies/failures, circuitry inadequacies and sensor location problems.

Propulsion Performance Margin Uncertainties. Propulsion performance margin determination requires input from a relatively large number of subsystems/disciplines. Some of the more significant ones are: on-board

propellant mass at liftoff, propellant temperature profiles, feedline pressure drops, pressurization system dispersions, engine operating dispersions, mission dispersions, planned emergency operating requirements and dispersions, minimum engine net positive suction pressure (NPSP) requirements, volume of unusable propellants (residuals), vehicle mass dispersions, vehicle aerodynamic drag loss dispersions, guidance loss dispersions, and low level cutoff sensor performance characteristics.

Of these parameters, onboard propellant mass at liftoff, propellant temperature profiles, feedline pressure drops, pressurization system performance, clustered engine operating performance, propellant residuals, and low level cutoff sensor characteristics are all verified during a propulsion system test program. Each of these subjects has been discussed in preceding paragraphs.

The risk of early inflight catastrophic failure is judged to be minor, while risk for mission loss is judged to be high. Mission loss risk may be reduced to minor by conservatively loading propellants for early flights. Instrumentation, in addition to the flight measuring program, and more flights may be required prior to achieving operational status or manning. A short duration FRF is of limited value, providing data on only two of six parameters relating to propulsion system performance.

Engine/Feed System Chill. An inadequately cooled engine and/or feed system usually results in engine failure to start and may result in engine damage. A booster type vehicle with engine start prior to vehicle liftoff can consider this a non-issue relative to flight catastrophe and mission loss risk. This is because the function is complete at engine ignition. The risk for launch delay, however, is high. Extensive chilldown development problems occurred during MPTA testing. Many development difficulties were also experienced on stages S-II and S-IVB of Saturn V and the S-IV stage of Saturn I—all three having engine altitude start requirements. For that reason flight catastrophe and mission loss risk are judged to be moderate for vehicles with engine start at altitude and high for vehicles with engine start in earth orbit.

System design depends heavily on propellants, engine design, and vehicle mission. Vehicles with engine start at altitude generally continue propellant chill systems operation beyond liftoff and until engines are started. Vehicles which start in earth orbit experience fluid management problems due to the reduced gravity environment in addition to engine/feedline chill problems which are more severe because hardware is warmer and may be filled with vapor. Analytical methods for defining engine/feedline chill are available but are of little value without basic data to anchor the methods. Necessary data can be achieved during propellant loading tests. The short duration FRF type tests provides actual proof of proper chill and is beneficial. For orbital start requirements, analytical methods are more complex and are even less dependable. Special ground tests are necessary including engine start. Extensive analyses are then required to prove engine start in the orbital environment is likely. Assurance of orbital start does not exist. An orbital experiment with demonstrated engine start was required for Saturn V S-IVB.

Testing, analytical methods, and correlation of analysis and test data require significant time, thus for some vehicle configurations and missions sole dependency on late testing at the launch site as the only source for data may prove to be exceedingly costly.

Tank Insulation. Flight catastrophe and mission loss risks are judged to be minor, while launch delay risk is high. A short FRF is beneficial but cannot be considered essential; however, more than one propellant loading test is essential to verify both thermal and structural performance. Instrumentation in excess of the normal flight measuring program is required to collect necessary data.

Insulation usage is dependent upon propellant type, mission, and vehicle configuration. Hydrogen filled tanks require insulation, while it may be optional for oxygen tanks. Vehicle design may determine if tanks with RP-1 require insulation. Insulated tanks filled with cryogenics often experience some insulation debonding—both small and large areas. Some factors affecting debonding are tank surface preparation, insulation type, quality control exercised in application processes, and tank deflections. For the early flight vehicles for programs without a propulsion system test program, insulation debond areas may be identified during propellant loading tests and repaired prior to flight. This can be highly ineffective considering launch schedule. The propulsion system test vehicle at a remote site provides for many loadings/firings relatively early in a development program and is an ideal method to improve insulation systems prior to flight vehicle delivery and exposure to cryogenic temperatures. Other necessary data from the loading tests are insulation system total heat leak which affects pressurization system design and ullage venting. Data obtained from tests require processing and extensive analysis prior to vehicle flight. Other items of interest are heat shorts (deficiencies in tank insulation) which may be conducive to liquid air/nitrogen formation—safety.

Analytical methods to predict insulation failures do not exist and possibly will never exist, thus testing is necessary. Heat input to propellants can be analytically determined but with significant inaccuracies, thus anchor points through testing are needed. If heat leak test data are unavailable, program continuation based on conservative designs are possible but can be costly in terms of weight, number of flights, and increased instrumentation.

Hardware Thermal Control. Flight catastrophe and mission loss risk are judged to be minor, however, launch delay risks are high. A short FRF type firing demonstrates all equipment functions at temperatures prevailing at engine start. The propellant loading test and standby periods preparatory to engine start are of immense value. Temperature sensitive propulsion, electronics, and structural hardware have temperatures determined during loading tests and compartment gas purges adjusted in quantity, temperature, and exhaust location to bring hardware within desired operational limits. Thermal insulation may be applied to components, as may heater blankets. These activities generally require considerable time, more than one propellant loading and instrumentation not normally planned for flight.

Vehicles with altitude start requirements may use such temperature data plus analyses to determine thermally acceptable temperatures are available at engine start. For missions requiring engine start in orbit, dependent upon elapsed time in orbit before engine start, component thermal conditioning pre-launch is of no value. Thermal analyses must be performed on all equipment, and those which are thermally unacceptable must be heated by heater blankets or cooled by some cold plate type system. Special testing independent of propulsion system testing may be required.

Stored Gas System. Flight catastrophe and mission loss risks are judged to be minor. Launch delay risk is minor, and launch complex risk is moderate. This is a high pressure system used to store gas which is needed by other systems/hardware. Test objectives are to ensure structural adequacy, that the charging procedure is compatible with vehicle/facility countdown and required onboard mass of gas, that the distribution system includes necessary redundancy, and that appropriate operational instrumentation is provided and is functional. Most aspects of the system can be well analyzed, however, some test data is beneficial for verification. Data from propellant loading tests verify most aspects except structural adequacy in an actual firing environment and adequacy of the gas mass. A short FRF will demonstrate environmental adequacy and limited information on gas mass.

APPENDIX 3

SPACE SHUTTLE MAIN PROPULSION SYSTEM

Shuttle Overview

The Space Shuttle is a reusable transportation system designed to transport people and equipment to earth orbit, to transfer payloads from one orbital position to another, and to return people and equipment to earth upon mission completion. Space Shuttle, propelled by a combination of liquid cryogenic propulsion and solid rocket propulsion systems, is comprised of three separate major elements which must function as a system. These elements are: an orbiter, which is the only element to reach earth orbit and includes personnel and cargo accommodations; an external tank containing necessary propellant for the liquid engines; and a solid rocket propulsive element providing lift capability during the initial two minutes of flight. Space Shuttle is shown on Figure 3-1.

Liquid cryogenic propulsion, the subject of this report, is provided by components located on the external tank (ET) and orbiter. Three Space Shuttle main engines (SSMEs) are located on the orbiter as are portions of propellant delivery systems, pressurization and loading systems, thrust vector control system, and major control elements. The ET contains hydrogen and oxygen propellants, propellant fluid system lines connecting the tankage to the orbiter, portions of pressurization and loading systems, and a minimum of controls and valves.

Liquid and solid propulsion systems are ignited prior to liftoff, and utilized in parallel during initial flight phases; both systems have nozzles which are gimbaled for thrust vector control. Solid rocket boosters (SRBs) are separated from the Shuttle approximately two minutes after liftoff, while the liquid system operates for an additional approximately six minutes. Prior to orbiter orbital insertion, the ET is separated from the vehicle. Neither liquid nor solid propellant systems are involved in orbiter return-to-earth.

The SRBs include parachutes for descent after separation and are recovered from the ocean for refurbishment and reuse on subsequent flights. The ET is expendable.

Liquid Propulsion System Overview

The orbiter-located SSMEs and liquid propulsion system hardware were designed for reuse. The expendable ET propulsion equipment was minimized and designed for minimizing cost.

An extensive data base containing analytical procedures, hardware design practices, manufacturing practices, and development techniques was established during the Apollo program. Since most of the information contained in this data base was directly applicable to the Space Shuttle design, non-engine Shuttle hardware and systems were designed maximizing the use of this information. The

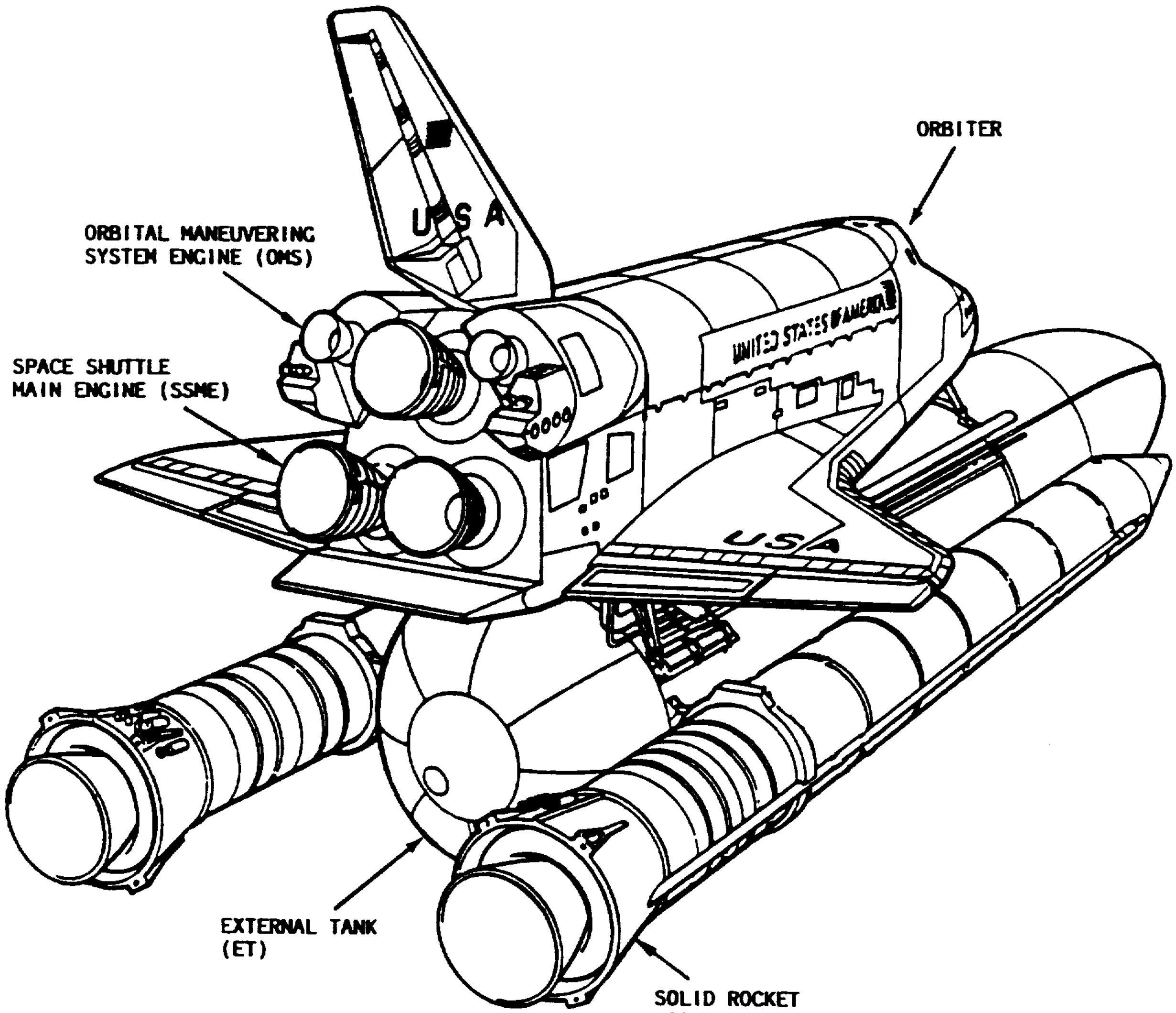

FIGURE 3-1. SPACE SHUTTLE

major new requirement imposed on the Space Shuttle was hardware reusability (55 flight capability). This requirement obviously necessitated some new technologies.

The SSME was an advanced concept, high-performance engine requiring innovations in design practices, materials, manufacturing, and inspections. Reuse requirements further complicated the development program.

The liquid propulsion system design for the Space Shuttle is shown schematically in Figure 3-2. Physical design of the ET and orbiter are shown in Figures 3-3 and 3-4, respectively, while Figure 3-5 shows the SSME fluid cycle. Approximately one and one half million pounds of liquid oxygen and one quarter million pounds of liquid hydrogen are carried by the ET. The three orbiter SSMEs are supplied with propellants from the ET through orbiter/ET disconnects located aft of the ET. SSME rated sea level thrust is 470,000 pounds, although the thrust is varied between 65 and 104 percent rated thrust during flight (109 percent in extreme emergencies).

Liquid Propulsion Subsystems

Liquid propulsion systems consist of numerous subsystems and many components which must function individually and collectively in an interactive environment. Major subsystems are described and discussed briefly in the following paragraphs.

Propellant Delivery System. Principal features of the propellant delivery system are:

- Fine mesh screens in each propellant tank at the entrance to each 17-inch diameter feedline

- ET located long oxygen and short "siphon" configuration hydrogen 17-inch diameter feedlines

- ET/orbiter interface 17-inch diameter quick disconnect and valving for each propellant

- Orbiter-located 17-inch diameter feedlines connecting quick disconnects and the respective propellant manifold

- Orbiter-located 12-inch diameter feedlines, three from each propellant manifold to 12-inch diameter prevalves–one prevalve per line–located immediately upstream of the inlet to each low pressure pump (six in all).

- Orbiter-located 12-inch diameter lines between the prevalves and low pressure pump inlets for each propellant (6 in all).

The 12-inch diameter feedlines have additional fine mesh screens located immediately upstream of each low pressure pump inlet to assure no engine

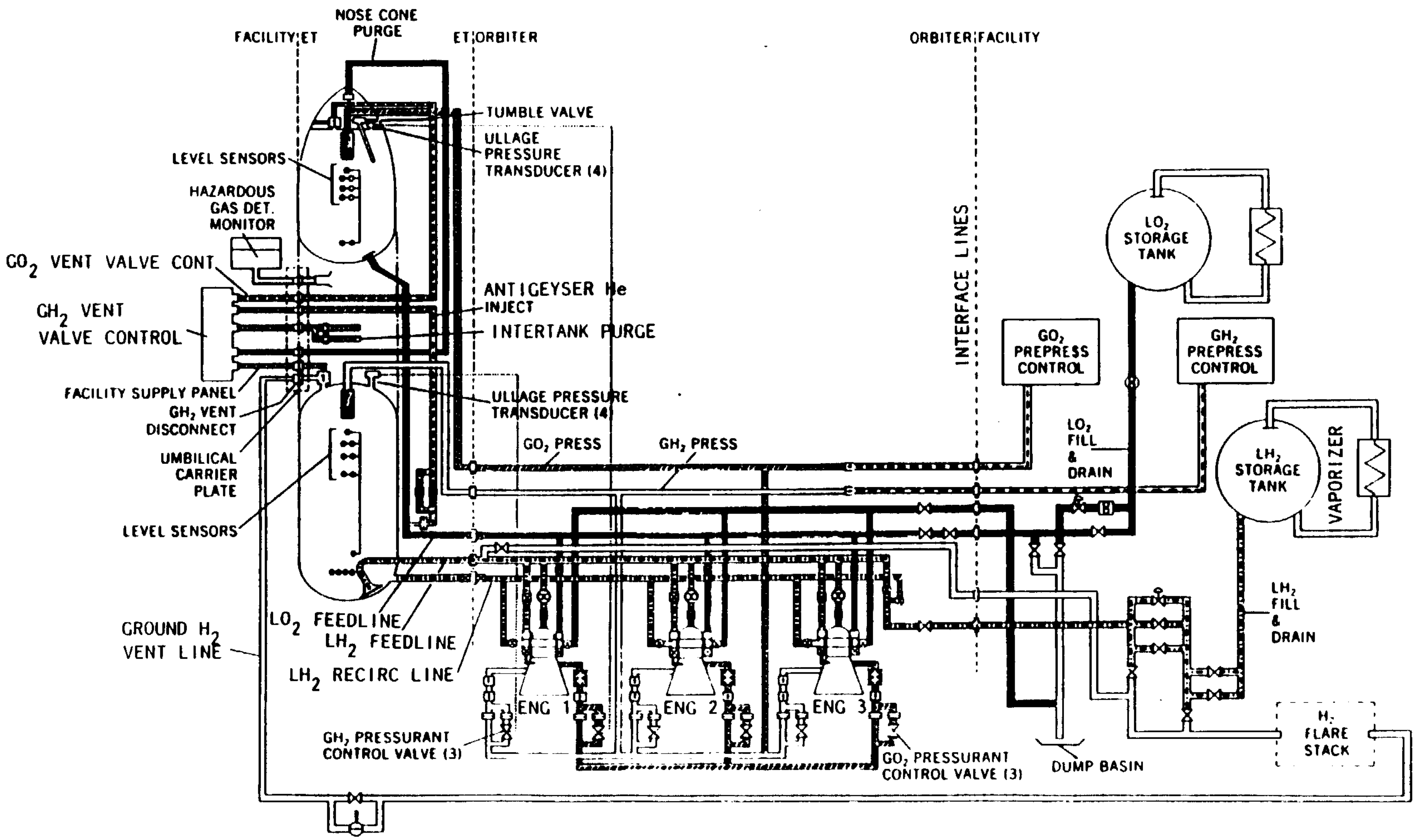

FIGURE 3-2. SPACE SHUTTLE MAIN PROPULSION SYSTEM SCHEMATIC

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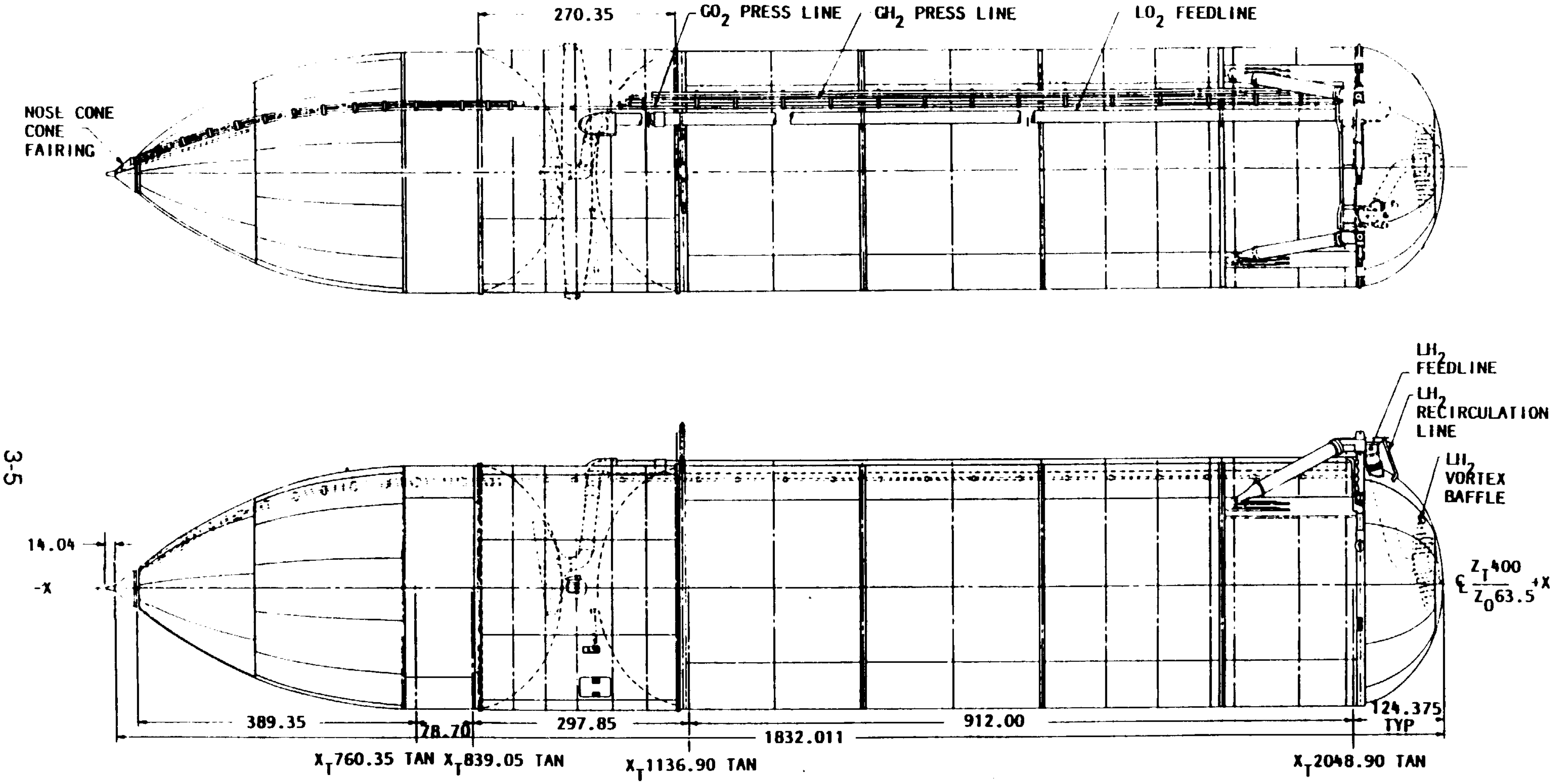

FIGURE 3-3. EXTERNAL TANK MPS HARDWARE

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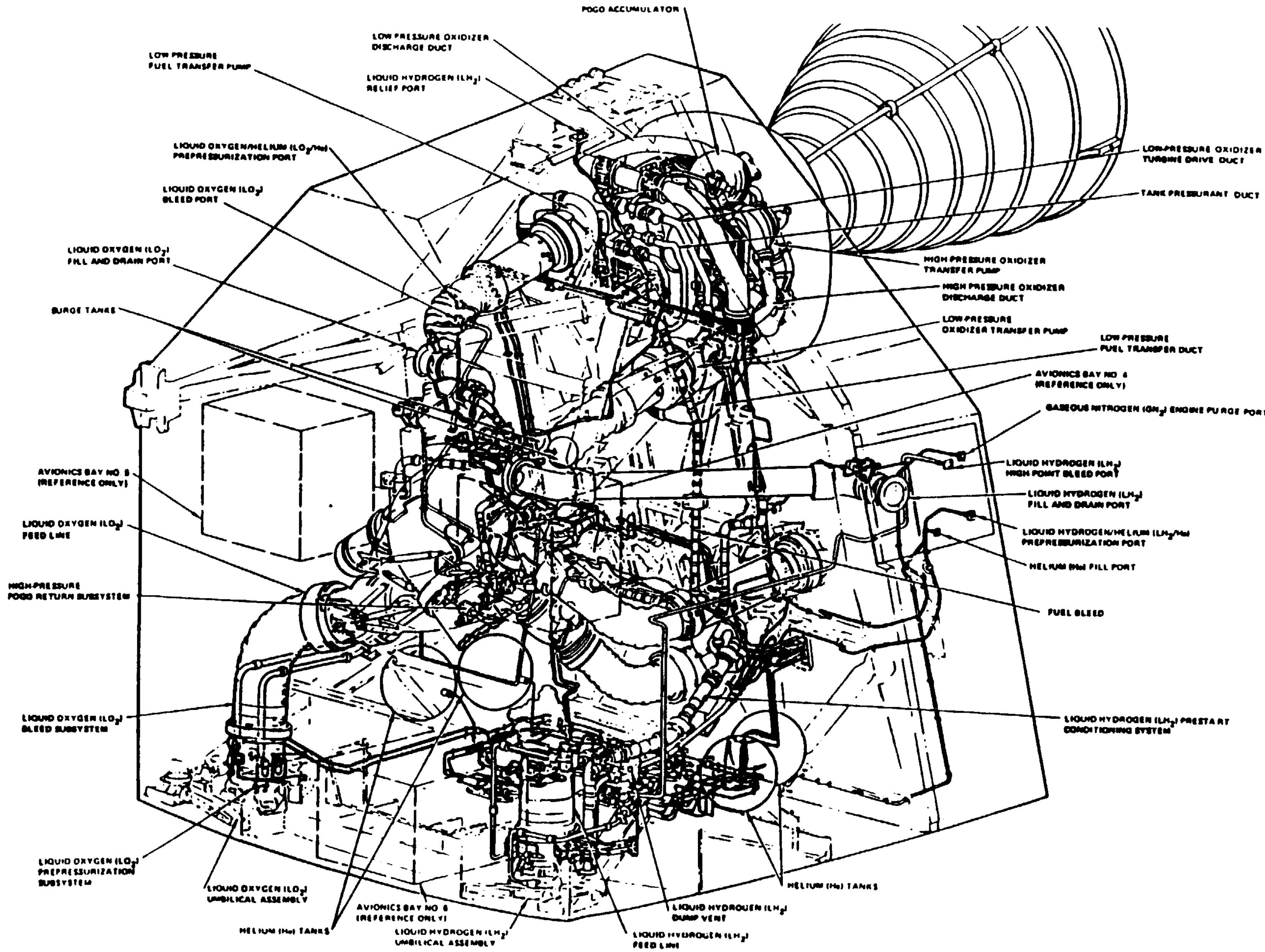
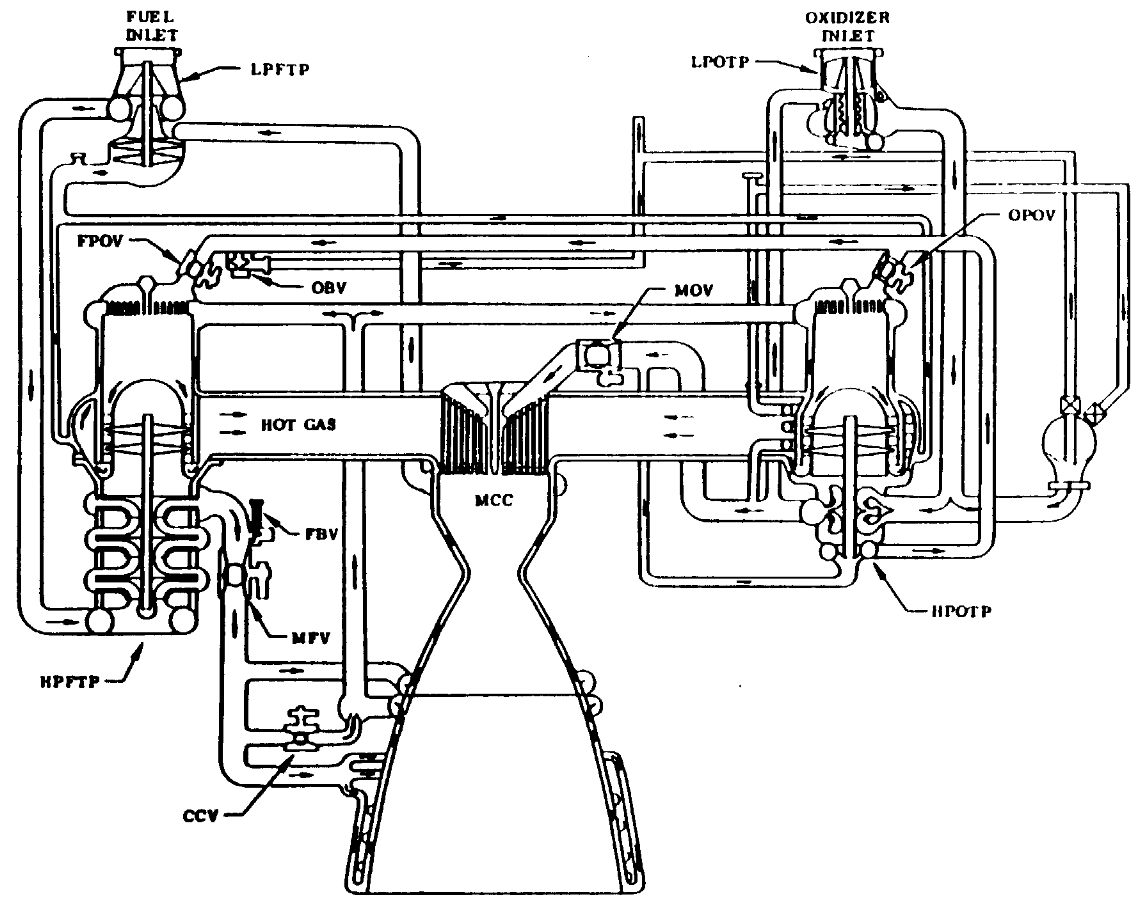


FIGURE 3-4. ORBITER AFT FUSELAGE
(MPS HARDWARE)

G 0034.01

C-2



LEGEND:

CCV	CHAMBER COOLANT VALVE	FPOV	FUEL PREBURNER OXIDIZER VALVE
MCC	MAIN COMBUSTION CHAMBER	OPOV	OXIDIZER PREBURNER OXIDIZER VALVE
FBV	FUEL BLEED VALVE	HPFTP	HIGH PRESSURE FUEL TURBOPUMP
OBV	OXIDIZER BLEED VALVE	HPOTP	HIGH PRESSURE OXIDIZER TURBOPUMP
MFV	MAIN FUEL VALVE	LPFTP	LOW PRESSURE FUEL TURBOPUMP
MOV	MAIN OXIDIZER VALVE	LPOTP	LOW PRESSURE OXIDIZER TURBOPUMP

FIGURE 3-5. SSME PROPELLANT SCHEMATIC

R-ENG 0035.01

damage from contamination. The feedlines use appropriate gimbal joints to assure required flexibility, and components use vacuum jacketed or foam insulation for thermal control purposes. One four-inch diameter line on the ET parallel with and connected to the oxygen supply line establishes a flow loop to maintain subcooled liquid oxygen in the feedline during prelaunch and prevents oxygen feedline geysering. An anti-geyser line, in which facility-supplied helium was injected to prevent flow interruptions, has subsequently been determined unnecessary and has been removed to save weight and cost although vehicle sensitivity to oxygen geysering is increased.

Hydrogen Feedline/Engine Thermal Conditioning

Acceptable engine start requires heat removal from rocket engine components so that sub-cooled propellants exist within the feedline at start command. The system accomplishing these objectives consists of: (1) a bypass line containing a valve and small electrically driven hydrogen pump about each hydrogen prevalve - one per engine; and (2) a hydrogen flow return line from each SSME hydrogen bleed valve to the ET hydrogen supply tank.

Return lines from each SSME bleed valve are manifolded prior to crossing the orbiter/ET interface through the quick disconnect enroute to the ET hydrogen tank. A high point bleed line—another significant feature of the system—is connected at the hydrogen feedline high point and terminates within the hydrogen ground vent line. Valving for controlling flow in this line opens during tanking and closes immediately prior to engine start. Hydrogen recirculation system operation commences during propellant loading and terminates immediately prior to engine start when prevalves are opened.

Performance verification objectives for this system are accomplished prior to engine ignition, and if properly conducted, do not entail significant program risk. While specific test objectives and related events from testing are reported in later sections of the report, it can be summarized here that testing has revealed marginal performance capability which necessitated procedural and limited hardware changes to achieve necessary performance.

Oxygen Feedline/Engine Thermal Conditioning

Requirements for oxygen hardware conditioning are similar to those for hydrogen, although the approach for accomplishing hardware conditioning is different. Prevalves and engine bleed valves are opened during propellant loading to permit a regulated flow quantity to continually flow through the engines. Flows exiting the SSMEs are ducted to ground systems for disposal. To assure acceptable propellant quality within the feedline at engine start, oxygen bleed flow from the ET is initiated only five minutes prior to start. This oxygen quantity from the tank is considered in the oxygen tank loading manifest.

Hydrogen/Oxygen Propellant Depletion Sensors

Engine protection from propellant depletion is necessary to assure safe engine shutoff. Sensors located within and near the bottom of the hydrogen tank and within the oxygen suction lines accomplish this purpose. Sensors are located to protect engines and minimize the unusable propellants. Multi-sensors are used for both hydrogen and oxygen.

Performance verification objectives are accomplished during hot firing and entail significant risks to engines and possibly vehicle and facility. Failure to satisfy required NPSP to engines for each propellant can damage the engine–possible pump explosion. Engine oxygen NPSP changes rapidly as the propellant tank is emptied. Insufficient hydrogen to an engine results in unacceptably high combustion gas temperatures which destroy hardware in milliseconds.

Propellant Delivery Dynamics

Interaction of vehicle feed system and engine flow disturbances with the vehicle structure can lead to devastating consequences. Closeness of the first/second mode structural frequencies to propulsion system frequency require inclusion of protective features–passive suppressors–which are integrated into the vehicle design. A suppressor is located between low and high pressure oxidizer pumps of each engine, charged initially with orbiter supplied helium, and with vaporized oxidizer from the heat exchangers during flight. Excessive suppressor pressurant is discharged into the single 17-inch diameter oxygen feedline near the orbiter/ET disconnect valve. Helium pressurant is substituted for oxidizer pressurant at shutdown. Care is exercised to assure the liquid gas interface of the suppressor is not broken. An approximate 25 percent shift in feedline frequency is attributed to the suppressor system.

Required data are obtained during hot firings, and processes for acquiring data entail some risks to the vehicle and facility. A hydraulic driven pulsing device installed on the oxygen feedline inputs pressure pulse to the fluid at varying frequency and amplitude. Accelerometers and pressure sensors, strategically located on feedlines and engine, measure responses which are then used with mathematics to predict structural stability characteristics of the vehicle. Other data are collected as appropriate to assess performance of designs which may be included to prevent the structural dynamic problem. Data are collected with varying propellant levels in tankage and with the suppressor system in both inactive and active modes. Numerous difficulties have been experienced in establishing testing conditions desired and in collecting data; however, these difficulties are largely non-flight hardware issues.

Propellant Tank Pressurization

SSME hydrogen and oxygen low pressure pumps require a minimum net positive suction pressure (NPSP) if acceptable start, operation and shutdown are to be achieved. Propellant tank ullage pressure is one of the controllable variables affecting NPSP. Ullage pressure is also needed for oxygen and hydrogen tank

structural stability during liftoff and flight. Systems have been designed to satisfy requirements for each propellant tank. Pressurization system principal features are:

- Pressurant gas sources
- Relatively small high pressure lines from pressurant sources to flow controlling modules
- Modules for controlling flow
- Lines from controlling modules to orbiter/ET quick disconnects and continuing lines to propellant tanks
- Gas diffusing system within propellant tanks
- Devices for sensing ullage pressures within tanks and for controlling flow control modules
- Vent/relief valves for tanks
- Ground-to-vehicle supply systems for pressurizing tanks

Hydrogen and oxygen tank pressurization systems are similar in design, although the gas sources differ. Heated gaseous hydrogen (liquid hydrogen tank pressurant) is available from each SSME nozzle wall coolant flow. Heated gaseous oxygen (liquid oxygen tank pressurant) is available from each engine-mounted heat exchanger. A ground storage supply provides ambient helium gas for both tanks during prefiring–approximately the final three minutes prior to engine start. Systems use gimbal joints and hardware insulation as appropriate to assure structural capability and hardware operativeness. Tank pressurization systems are shown in Figures 3-6 and 3-7.

Many specific objectives are included in system verification. The paramount objective, however, for the pre-firing pre-pressurization system and the pressurization system active during firings, is to maintain tank ullage pressures in both hydrogen and oxygen tanks within specified control bands for all operational phases and potential failure modes. The original designed system has met requirements, although numerous problems have occurred relative to control module orifice sizing, control valve sluggishness in operations, oxygen heat exchanger internal coil pressures for some conditions less than the fuel rich hot gas flow external to coils, software changes relating to pre-pressurization control and other areas.

Propellant Loading

The Space Shuttle propellant delivery systems, previously described, are used for both hydrogen and oxygen propellant loading. Additional required hardware for each propellant is limited to: (1) a 12-inch diameter line from the

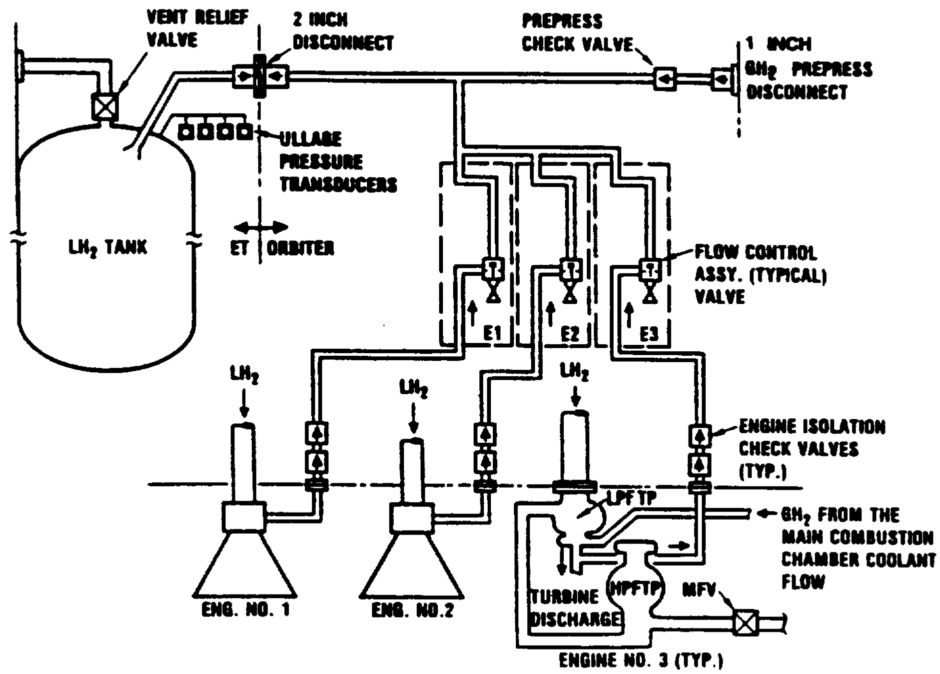


FIGURE 3-6. GH_2 PRESSURIZATION SYSTEM SCHEMATIC

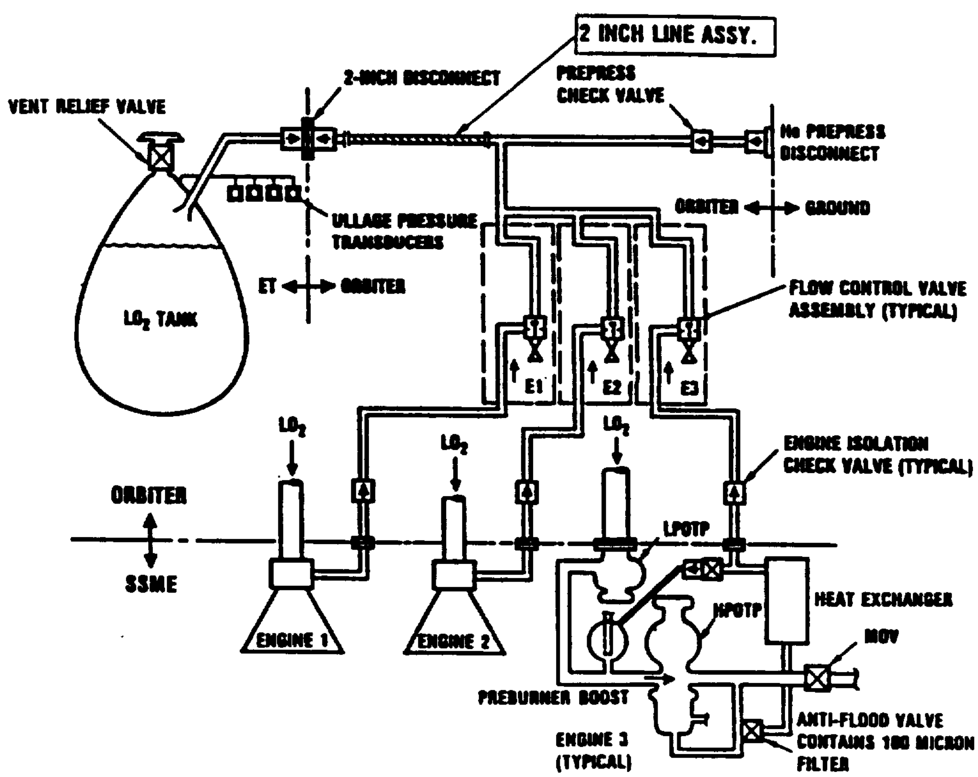


FIGURE 3-7. GO_2 PRESSURIZATION SYSTEM SCHEMATIC

orbiter-located manifold to the vehicle skin; (2) two series valves in this line; and (3) an orbiter-to-ground quick disconnect valve. Critical to the loading process is operational instrumentation - liquid level sensors within tanks for managing operations; selective temperatures and pressures measured within tankage, vent systems, propellant delivery systems, and ground located hardware for fluid condition monitoring.

Aside from undesirable/unsafe aspects of hydrogen leaks during loading, the physical process of transferring hydrogen from ground systems to the Space Shuttle is simple relative to oxygen loading. Vehicle hardware damage attributable to slugging oxygen flow during facility/vehicle hardware chilldown is a possibility. Fluid geysering and resultant serious damage to hardware is possible once oxygen commences collecting in facility vertical lines enroute to the vehicle and within engine and vehicle lines. Thorough procedures are essential for these operations. Many procedures developed prior to testing have been amended; some were totally rewritten, based on new information, and all were verified during the many months of testing. The many test requirements are defined in Table 3-1 and cover all aspects of normal and contingency operations.

Vehicle hardware used for propellant loading is used also for propellant removal.

Propellant Loading At Elevated Tank Pressure

ET design philosophy was based on: (1) hydrogen and oxygen tanks being capable of "free standing" on the launch pad and; (2) no required internal pressure during propellant loading and standby operations with propellant aboard. This philosophy changed during the propulsion system test program, and requirements, which varied with time, were imposed for internal tank pressures during propellant loading and some standby operations prior to and after firings. Vehicle vent valve cycling to control internal tank ullage pressure within specified limits during propellant loading and post loading operations was investigated, selected as the preferred approach and demonstrated.

Other requirements restricting propellant levels during loading to satisfy hardware thermal limits were also imposed. Such requirements were managed by procedures.

Late recognition of these requirements necessitated total development of concept, necessary engineering, and procedures after system testing had commenced. Primary concern relative to control method implementation was tank overstress and potential blowup. Considerations of both normal and contingency operations were required for not only ullage pressure control but for other disciplines such as loading to assure geysering did not occur.

Propellant Loading Instrumentation and Software

SSME oxidizer-to-fuel consumption is 6:1 and has been essentially constant during missions. Propellant quantities loaded were thus based on engine

consumption requirements plus performance reserves and unusable propellants. To facilitate loading to specified requirements, propellant level sensors are located at designated locations in each tank. Multiple sensors are located at the designated 100 percent propellant level and other levels in close proximity. Sensors are also located within tanks to provide propellant levels during loading and prevent tank overfill.

Flight mass is achieved when monitored sensor/sensors register wet a specified percent of the time–15 to 75 percent–continuously under specified conditions. Loading activity at KSC is computer controlled and major parts of software development are critically dependent upon propellant loading sensor data collected during propulsion system testing.

This complex area resulted in many surprising findings, thus test requirement changes were necessary as events unfolded. Data collection time exceeded that for any test requirement. Unexpected propellant flow currents in tankage adversely affected sensors and required sensor shielding. Initially available electronics for automated loading control were inadequate and required replacement; software changes for hardware compatibility were essential; and necessary input data were collected during hours of testing devoted to this one subject.

Oxygen/Hydrogen Feedline Safings

Propellant delivery systems on the orbiter plus parts of the fill-and-drain systems (both hydrogen and oxygen) are filled with propellants. Hardware safing after main engines shutoff, and ET/orbiter separation is necessary. Hydrogen fill-and-drain valves and prevalves are opened, and the system is vented to vacuum. The oxygen prevalve and main engine oxygen valve are opened and the oxygen system vented to vacuum through engine nozzles. Both systems are then purged with helium from vehicle storage bottles. Appropriate valving is then closed preparatory to orbiter reentry at some later date. Test objectives for this issue and the following issue, return-to-launch-site (RTLS) dump, are not unique but were necessary to verify prepared procedures with particular emphasis on the time line.

RTLS Dump System

An aborted Shuttle mission with the orbiter returning to the launch site results in propellant delivery systems full of propellants. Disposal of hydrogen is most important prior to re-entering earth's atmosphere. Opening RTLS valves permits hydrogen propellant to escape. System purging with ambient temperature helium further safes the system.

Aft Compartment Safing

The aft compartment of the orbiter, between bulkhead 1307 and the aft heat shield, contains three SSMEs (excluding nozzles), total propellant delivery systems located on the orbiter, propellant fill-and-drain systems, SSME control electronics, vehicle propulsion system electronics/instrumentation, other propulsion system equipment, and a gas distribution system for inerting and thermal conditioning

compartment and equipment. Many low and high pressure pipe connections for both oxygen and hydrogen systems are sources for leaks.

The compartment inerting and thermal control system consists of numerous nozzles for distributing a ground supplied heated nitrogen flow. Gas sensing probes within the compartment alternately provide compartment gas samples for real-time analysis and test operation decisions regarding safety. Strategically located vacuum sample bottles throughout the compartment obtain gas samples during simulated flight phases for postflight analysis. For safety enhancement, propulsion system test hardware uses non-heated nitrogen purges of significantly greater quantity than the heated nitrogen purge used on both the flight and test article. Also, fast response hydrogen sampling devices are located within the aft compartment of the test article, as are fire detection sensors. Other compartments of the vehicle such as the ET intertank use somewhat similar inerting, thermal control and monitoring systems.

Miscellaneous Vehicle Systems

Numerous systems such as the hydraulic system for vehicle control and systems for tank inerting prior to propellant loading have not been discussed. Proper operation of each system collectively with other systems is essential for success. Previous discussions of systems are believed to suffice for portraying insight into Space Shuttle and the propulsion system test program.

Space Shuttle Main Engine

The SSME is a high performance, high technology engine with reuse capability. None of these characteristics impose significant complexity to design or operating features of the vehicle propulsion system. It merely implies a compact, high performance thrust package for vehicle use. Such an engine design, however, generally requires longer development time and more testing to reach maturity, and if used prior to achieving maturity, vehicle risks occur and test program delays are likely. The SSME was not adequately mature for approximately the first half of the propulsion system test program.

The SSME uses a staged combustion cycle in which propellants are partially burned at low mixture ratio, high pressure, and relatively low temperature in the preburners and then completely burned in the main combustion chamber. The SSME flow schematic is shown in Figure 3-5. The propellant system uses four turbopumps. The two low pressure turbopumps operate at relatively low NPSP to permit low pressures in the vehicle tanks. The function of these pumps is to provide sufficient pressure at the inlets of the high pressure turbopumps to permit them to operate at high speeds. Approximately 75 percent of the flow from the high pressure oxidizer turbopump (HPOTP) goes directly to the main combustion chamber. Approximately 10 percent is directed to the preburner pump, which raises the pressure to that required by the preburners. Small quantities are bled through the heat exchanger for oxidizer tank pressurization and POGO suppression pressure. The balance of the oxidizer drives the turbine powering the

low pressure oxidizer turbopump (LPOTP) and is then recirculated to the inlet of the HPOTP.

Approximately 20 percent of the high pressure fuel turbopump (HPFTP) discharge flow is used to cool the main combustion chamber (MCC), drive the low pressure fuel turbopump turbine, cool the hot-gas manifold and injector, and provide fuel tank pressurant. The remaining fuel is first used to cool the nozzle, then supply the preburners.

The hydrogen-rich steam generated by the fuel and oxidizer preburners first drives the high-pressure pump turbines, then flows to the main injector where it is mixed with additional oxidizer and fuel and injected into the main combustion chamber.

The main injector is baffled, coaxial-element injector, having dual faceplates that are transpiration-cooled by gaseous hydrogen. The gimbal bearing is bolted to the main injector and dome assembly and is the thrust interface between the engine and vehicle. It also enables the engine to be gimballed to provide pitch, yaw, and roll control. The MCC consists of an internal coolant liner and an external structural jacket. The main combustion chamber operates near 3300 PSI and experiences exceedingly high wall heat transfer rates. Thus to avoid burn through, a unique design is necessary. To enhance heat transfer, the chamber is made of copper alloy which is cooled regeneratively with hydrogen flowing through milled axial coolant channels. The nozzle is bolted to the MCC and is constructed of tapered tubes reinforced by a structural jacket with insulated hatbands. The tubes provide an up-pass fuel coolant circuit that is supplied with hydrogen through three down-pass transfer ducts connected to the aft inlet manifold. The nozzle has an expansion ratio of 77.5, although during development, two engine nozzle configurations were utilized–flight nozzle configuration with large area ratio and non-flight configuration with much smaller area ratio.

Commands to and within engines are controlled by a separate controller for each engine. To work effectively as an engine system and as a subsystem within Space Shuttle, other significant engine components are required as control valves, sensors of all types (temperature, pressure, accelerometers), hydraulic actuators, and ignition systems.

SHUTTLE PROPULSION SYSTEM TEST OBJECTIVES

Propulsion system test objectives were initially developed to assure inclusion of capabilities essential for satisfying vehicle, engine, facility, and GSE needs. The broad test objectives follow:

- Determine structural acceptability of engines in the propulsion system test environment

- Determine functional capability of engines in the propulsion system test environment

- Validate engine-imposed vehicle design requirements
- Determine engine propellant leakage integrity
- Validate structural integrity of major elements (ET, Orbiter, SRB) and the interactive loads between major elements
- Validate structural integrity of propulsion systems of individual elements to the interacting environments
- Validate the functional capabilities of propulsion system components/subsystems in the propulsion system environment
- Validate the functional capability of electrical hardware, computer control system/software with vehicle/propulsion system environments
- Validate vehicle/propulsion system imposed engine requirements
- Determine propulsion system leak integrity
- Determine vehicle external design environments
- Determine functional compatibility of the vehicle structure, fluid, electrical, hydraulic, and software systems with the ground/GSE interfacing systems.

These broad objectives were amended as appropriate and specific test requirements to be satisfied during propulsion system testing were developed. Table 3-1 is a total list (approximately 300) of test requirements, including initial requirements and others added throughout the program. The initial requirements, approximately 200, were carefully integrated and scheduled to form a test program. Twelve hot firings of varying time durations from a few seconds to approximately 550 seconds, one non-firing propellant loading test, and two structural tests (resonant survey) were identified to be necessary.

Composition of each of the twelve hot firings actually conducted during propulsion system testing bears little resemblance to initially planned tests because: (1) hardware available for use largely controlled specific test requirements; (2) testing delays created immediate data needs to be satisfied wherever possible to minimize program impacts and; (3) test requirements were added to the program. Although test requirements were altered (shifted) significantly between tests, each test requirement of Table 3-1 was totally or partially satisfied (all satisfied to an acceptable level) in the propulsion system test program.

Additional test requirements, which were identified as testing progressed and integrated into the program, were in many cases crucial to Space Shuttle development/verification. These test requirements, involving both hot firings and non-hot firing type tests, resulted from oversight during the initial planning period, design changes to the vehicle/facility from earlier periods, failure of

TABLE 3-1. SPACE SHUTTLE PROPULSION SYSTEM TEST REQUIREMENTS

(TRSD PARAGRAPH TITLES AND NUMBERS - AMENDED)

TDS 5.1.1 TVC Performance Evaluation (12)
1. Sinusoidal Engine Gimballing
- 0.2° Amplitude
- 0.4 degrees amplitude
- 0.6 degrees amplitude
2. Step Response
3. Ramp Response
4. Stroking Response
5. Flight Profile
6. Simultaneous Engine Gimballing/Throttling
7. Routine Engine Positioning/SSME Side Load Evaluation
8. Engine Clearance Checks (Non-Firing)
9. Hydraulic Power Failure Simulation (Non-Firing)
10. TVC Channel Failure Simulation
11. Halting Ramp Response
12. Simultaneous Gimballing and POGO
TDS 5.1.2 Hydraulic System Performance Verification (3)
1. Evaluate Hydraulic System during Engine
Throttling and Steady State Engine Performance
2. Hydraulic Warmant Flow Evaluation at Constant Return
Pressure, Constant Supply Temperature, and
KSC Countdown Timeline
TDS 5.2.1 LO2 Flight Pressurization System Performance (15)
1. Demonstrate satisfactory transition from
pressurization to autogenous system.
2. Verify pressurization system performance at RPL
with three engines.
3. Verify system performance at FPL with three engines.
4. Verify system performance at MPL with three engines.
5. Verify system performance with two engines at RPL.
6. Verify system performance with two engines at FPL.
7. Verify system performance with two engines at MPL.
8. Verify system performance with one control valve
failed open.
9. Verify system performance with one control valve
failed closed.
10. Determine pressure control with one engine out
during engine throttling, including effect of one
engine cutoff transient.
11. Determine pressure control with all engines being
throttled.
12. Verify pressurization system will support SSME's
during LO2 depletion cutoff.
13. Switch to spare ullage pressure transducer after
ET is pressurized.
14. Simulate RTLS or AOA with an engine out for at
least 300 seconds.
15. Optimize flow control valve orifice size.

Note: TRSD - Test Requirement & Specification Document
(XX) - Number if Original Requirements per TDS.

TABLE 3-1. SPACE SHUTTLE PROPULSION SYSTEM TEST REQUIREMENTS (CONTINUED)

TDS 5.2.2 LO2 Prepressurization System Performance (9) Verification
1. Verify capability to prepress within allotted time period.
2. Verify capability to maintain pressure within required band using operational launch countdown.
3. Verify capability to maintain pressure within required band for countdown hold.
4. Demonstrate that system will support drain of fully loaded ET.
5. Demonstrate that system will support pre- and post-tanking purge operations.
6. Verify that the heated helium purge in pressurization line prevents ice and frost formation on line.
7. Evaluate LO2 tank ullage back pressure.
8. Evaluate ET vent valve relief performance.
9. Demonstrate that system will support 5000 gpm drain rate.
TDS 5.2.3 Propellant Conditioning - LH2 Recirculation (12) Performance
1. Evaluate MPS-SSME Prestart LH2 Conditioning
2. Demonstrate ability to meet SSME start requirements during operational timeline.
3. Validate system heat leak design analysis.
4. Evaluate operation of LH2 high point bleed.
5. Determine time/sequence requirements for initiation of LH2 recirculation.
6. Determine effects of propellant loading operations on recirculation system.
7. Demonstrate compatibility of recirculation cutoff-SSME start sequencing.
8. Evaluate effects of countdown holds.
9. Evaluate effects of countdown recycles.
10. Demonstrate procedure for reinitiating recirculation operation following termination.
11. Evaluate 2 minutes launch hold.
12. LH2 Recirc Manifold Rel. Vlv. Check
TDS 5.2.4 Pneumatic System Performance (16)
1. Demonstrate GSE sequence for filling and maintaining helium bottles.
2. Evaluate helium storage temperature/pressure profiles during tanking/prelaunch operations.
3. Determine effects of countdown holds.
4. Determine effects of countdown recycles.
5. Determine optimum pressurization sequencing.
6. Demonstrate system capability for OFT Mission and extended boost duration.
7. Demonstrate ability to operate Orbiter valves for worst case post-MECO conditions.
8. Demonstrate LO2/LH2 feedline repressurization.

TABLE 3-1. SPACE SHUTTLE PROPULSION SYSTEM TEST REQUIREMENTS (CONTINUED)

TDS 5.2.4 (Continued)
9. Support LO2 dump through SSME's.
10. Demonstrate RTLS dump/purge operation.
11. Evaluate capability to distribute helium to the SSME's.
TDS 5.2.5 Propellant Conditioning - LO2 Bleed (11)
1. Same as H2 discussed previous except for following:
2. Determine bleed system chilldown/sequencing requirements.
3. Determine optimum LO2 bleed flowrate.
4. Evaluate POGO recirculation line conditioning
5. Verify bleed system compatibility with POGO recirculation flow.
6. Evaluate effects of LO2 drainback on bleed system performance.
7. Evaluate Effects of Antigeyser line removal.
TDS 5.2.6 MPS Cluster Performance (29)
1. Verify MPS design compatibility with Orbiter structural heat shield and engine mounted heat shield.
2. Verify three-engine simultaneous start capability.
3. Evaluate MPS configuration effect on SSME operations.
4. Evaluate SSME start/shutdown transient variations.
5. Verify SSME operation at power levels from 65 to 109%.
6. Verify satisfactory MPS performance with engine out.
7. Demonstrate LH2 depletion SSME shutdown.
8. Demonstrate seven-hour unpressurized hold condition.
9. Demonstrate two-minute pressurized hold condition.
10. Demonstrate SSME POGO suppression system effectiveness.
11. Demonstrate effects of Engine gimballing.
12. Demonstrate ET/Orbiter/SSME interface compatibility.
13. Demonstrate capability to operate for at least 300 seconds with engine out.
14. Demonstrate LO2 depletion SSME shutdown from 65% power level, at varying depletion timer settings.
TDS 5.2.7 LH2 Flight Pressurization System Performance(16)
1. Same as for O2 system previously covered.
TDS 5.2.8 LO2 Propellant Feed System (20)
1. Evaluate start transients to SSME RPL.
2. Evaluate start transients for simultaneous three-Engine start.
3. Evaluate steady-state operation for three SSME operations at various power levels.
4. Evaluate steady-state operation with upper and lower engine at various power levels.

TABLE 3-1. SPACE SHUTTLE PROPULSION SYSTEM TEST REQUIREMENTS (CONTINUED)

5.2.8 (Continued)
5. Evaluate throttling transients:
6. Demonstrate effects of POGO oscillations
7. Evaluate effects of early engine cutoff
8. Validate ET/Orbiter and SSME/Orbiter Fluid Mechanical Interface.
9. Evaluate engine cutoff transients.
10. Validate prevalve closing sequence at MECO.
11. Validate ECO sensor system, cutoff delay, residules.
12. Verify prevalve relief function with no close and closed commands.
TDS 5.2.9 LH2 Propellant Feed System (19)
1. Same as for O2 previously covered.
TDS 5.2.10 LH2 Prepressurization System Performance (7) Verification
1. Same as discussed earlier for O2.
TDS 5.2.11 LH2 Propellant Loading/Detanking Procedures (19)
1. Verify drain capability for propellant trapped between the inboard/outboard fill and drain valve.
2. Determine ground interface pressure requirements for propellant chilldown, fast fill, and topping.
3. Verify MPS/Orb design prevents liquid air/N2 formation.
4. Verify capability to re-initiate loading
5. Verify capability of sequential/simultaneous loading/draining.
6. Evaluate LH2 tank loading accuracy.
7. Evaluate LH2 screen filtration/structural performance.
8. Evaluate propellant load/drain time requirements.
9. Evaluate LH2 point sensor system performance
10. Determine LH2 replenish rates.
11. Evaluate GSE capability to monitor/control propellant loading/draining.
Develop loading procedures compatible with Space Shuttle launch constraints for:
12. - Two-hour operational launch countdown. (75 min)
13. - Four-hour launch countdown. (165 min)
14. Evaluate LH2 Orbiter/GSE carrier plate purge effectiveness.
15. Determine effect of seven-hour hold with LH2 tank unpressurized
16. Determine effect of two-minute hold with LH2 tank pressurized.
17. Determine MPS/ET LH2 vent system back pressure under cryogenic gas flow during chilldown, fast fill, and replenish.
18. Verify relief capability of inboard F&D valve.
19. Perform a countdown recycle from T-4.3 sec. and replenish LH2 to 100%.

TABLE 3-1. SPACE SHUTTLE PROPULSION SYSTEM TEST REQUIREMENTS (CONTINUED)

TDS 5.2.12 LO2 Propellant Loading and Detanking Procedures (23)
1. Same as for H2 previously discussed except as follows.
2. Determine effect of helium bubbling on LO2 loading.
3. Evaluate effectiveness of LO2 anti-geyser system.
4. Evaluate adjustments of engine LO2 variable bleed.
5. Evaluate effects of antigeyser line removal.
6. Evaluate 5000 gpm drain
7. Optimize vent orifice size.
8. Evaluate 1300 GPM Loading with Vent Valve Cycling
9. Demonstrate 5 minute Pressurized Hold Capability
TDS 5.2.13 Engine GN2 Purge Verification (2)
1. Demonstrate MPS GN2 purge supply from the GSE meets ICD requirements
2. Verify proper control and operation of GN2 purges during countdown sequence.
TDS 5.2.14 ET/Orbiter/SSME Propellant Systems Purge and Inerting (3)
1. Develop procedure to purge the propellant systems prior to propellant loading.
2. Develop integrated Orbiter/ET/SSME MPS purging/inerting procedure which minimize the quality of purge gas and number of samples required.
3. Develop a procedure for purging of the Orbiter/SSME MPS while isolated from the ET.
TDS 5.2.15 Pneumatic Launch Support GSE Verification (7)
1. Verify capability of GSE to prepressurize the LO2 and LH2 tanks with ambient helium.
2. Evaluate capability of GSE to fill/maintain the onboard helium spheres.
3. Evaluate GSE monitor/control functions during propellant loading/draining.
4. Evaluate integrated pressurization system control during countdown operation.
5. Demonstrate capability of SSME GN2 purge panel.
6. Demonstrate capability of LH2 and LO2 pressurization panel.
7. Demonstrate capability of helium pressure reduction and bottle fill panel.
TDS 5.2.16 LO2/LH2 ET Self-Pressurization and Boiloff (2)
1. Evaluate the LO2 and LH2 Et vent valve relief operation during self-pressurization.
2. Determine LO2 and LH2 boiloff rates with tanks loaded to 100% and vent valves open.
TDS 5.2.17 LH2 RTLS Dump Verification and line relief valve
TDS 5.2.18 Verify LO2 Dump through the SSME's and feed line relief valve.

TABLE 3-1. SPACE SHUTTLE PROPULSION SYSTEM TEST REQUIREMENTS (CONTINUED)

TDS 5.2.19 LO2 Antigeyser System Integrated Performance (2)
1. Verify LO2 antigeyser system performance over full range of replenish and bleed flow rates, replenish temperatures, and helium supply pressures.
2. Verify adequacy of system operating procedures and determine hold times for loss of helium bubbling during replenish.
TDS 5.2.20 POGO Suppression System Verification (2)
1. Verify POGO accumulators remain charged with gas.
2. Verify POGO suppression system can vent the required flow without affecting engine performance, LO2 depletion sensors and other engines in case of early shutdown.
TDS 5.3.1 Maintainability Evaluation
TDS 5.3.2 ET and Orbiter T-0 Umbilical Motion Measurement due to cryogenic loading.
TDS 5.4.1 Engine Firing Thermal Soakback Determination (3)
1. Obtain Heat Shield Temperature Profile.
2. Determine engine firing thermal soakback effects.
3. Determine MPTA component temperature profiles.
TDS 5.4.2 Aft Fuselage Thermal Study (5)
1. Evaluate helium sphere pressures and temperatures during operational sequencing.
2. Evaluate aft fuselage purge effectiveness.
3. Evaluate propellant feed system heat loads.
4. Determine SSME thermal characteristics during chilldown and propellant loading.
5. Evaluate MPTA component temperatures during all operational phases.
TDS 5.4.3 Determine Hyd Actuation System environment
TDS 5.4.4 SSME Plume-Induced Environment and SSME (2) Thermal Performance Verification.
1. Evaluate thermal environment and thermal response of SSME nozzle, hat bands, and thermal protection material, turbine seal drainline, and manifold for all operational modes.
2. Obtain data to validate the plume thermal model.
TDS 5.5.1 Aft Fuselage Primary and Thrust Structure (3) Dynamic Stress and Load Transmission Evaluation.
1. Evaluate aft fuselage structural loads.
2. Determine SSME ignition overpressure effects.
3. Evaluate LPOTP inlet housing strains on Engines.

TABLE 3-1. SPACE SHUTTLE PROPULSION SYSTEM TEST REQUIREMENTS (CONTINUED)

TDS 5.5.3 MPT POGO Pulsing Evaluation (1)
1. Evaluate POGO characteristics of integrated propulsion system by pulsing the LO2 feed system: With and without suppressours.
TDS 5.5.4 Vibro-Acoustic Environment (1)
1. Evaluate aft fuselage vibro-acoustic environment With Vibro-Acoustic Water before Ignition
TDS 5.5.6 Perform MPT Resonant Modal Vibration Survey Test (1)
TDS 5.6.1 EIU Evaluation (1)
1. Evaluate capability of engine interface unit to control and monitor the SSME's during firing and to reprogram the SSME controller memory.
TDS 5.6.2 FC-MDM and Flight Instrumentation Evaluation (1)
1. Evaluate flight instrumentation (DFI, OI) function performance, compatibility with other avionic elements and functional performance of FC-MDM in controlling SSME gimballing.
TDS 5.6.3 Load Control Assembly/Power Control Assembly Operation Analysis - EPD&C Evaluation (1)
1. Evaluate compatibility of power control and load control assemblies with interfacing subsystem components.
TDS 5.7.1 Compartment Positive Pressure/Purge Flow Test(2)
TDS 5.7.2 Purge/HGDS Operations during Tanking/Static Firing. (1)
1. - Evaluate ET/Orbiter disconnect plate cavities and frangible nut canisters.
TDS 5.7.3 Aft Fuselage Purge Flow Balance and Verification Test. (1)
TDS 5.7.4 HGDS Simulated Inerting Test with simulated leakage.
TDS 5.7.5 ET/Orbiter LH2 and LO2 Disconnect Purge Leakage nd Flow Balance Verification Test (1)
1. LO2 and LH2 Disconnect - Ambient Conditions: Verify minimum plate gap pressure, flows out of frangible nut canisters and plate gap cavity outlets.
TDS 5.8.1 Transportability and Handling Demonstration
TDS 5.8.2 Checkout and Processing Demonstration - ET
TDS 5.8.3 Intertank Leak Measurement Verification

TABLE 3-1. SPACE SHUTTLE PROPULSION SYSTEM TEST REQUIREMENTS (CONTINUED)

TDS 5.8.4 Antigeyser Performance with and without Antigeyserline and LO2 Screen Evaluation
TDS 5.8.5 Venting Performance and LO2/LH2 Boiloff Rate Evaluation (2) 1. Verify performance of ET vent system during propellant loading, topping, replenish, pre-pressurization, and detanking. 2. Validate method of monitoring LO2 ullage pressure
TDS 5.8.6 Intertank Purge Verification (1) 1. Evaluate intertank purge effectiveness, intertank environmental thermal model and ET intertank temperature fix.
TDS 5.8.7 ET Propellant Lines Structural Verification -(1) Evaluate line capability to withstand loads induced during purging, propellant loading/unloading, and static firing.
TDS 5.8.8 LO2 and LH2 Tank Diffuser Qualification
TDS 5.8.9 LH2 Stratification Model Validation
TDS 5.8.10 ET Electrical/Instrumentation System Performance
TDS 5.8.11 ET Thermal Protection System Performance Verification.
TDS 5.8.13 LO2 and LH2 Tank Structure Thermal Model validation.
TDS 5.8.14 LO2 and LH2 Tank Thermal Protection System Surface Temperature
TDS 5.8.15 Intertank Structural Temperatures Thermal Model Validation.
TDS 5.8.16 Intertank "Y" Joint nitrogen Condensation Investigation.
TDS 5.8.17 Intertank Equipment Panel Thermal Performance
TDS 5.8.19 LH2 Tank Ullage Gas Thermal Model Validation
TDS 5.8.22 LO2 Feedline and Antigeyser Line Thermal Protection System Performance Validation
TDS 5.8.23 GH2 Vent Line Thermal Protection System Performance Validation

TABLE 3-1. SPACE SHUTTLE PROPULSION SYSTEM TEST REQUIREMENTS (CONCLUDED)

TDS 5.8.24 Pressurization Line Mounting Heat Leak Determination
TDS 5.8.25 Structural Dynamic Performance Validation - ET
TDS 5.8.26 ET Nose cap GN2 purge
TDS 5.8.27 GH2 Vent and Relief Valve Sense Line Purge evaluation.
TDS 5.8.28 Maintainability Evaluation - LRU and replacement retest
TDS 5.8.29 ET Ice Formation Assessment
TDS 5.8.30 RSS Thermal Model Simulation Test
TDS 5.8.31 Verify capability to maintain LH2 Tank Pressurized during fill.
GENERAL - -NO SPECIFIC TDS COVERAGE:
1. Verify vehicle flight test data evaluation techniques by assessing MPTA test data
2. Provide training for launch site ground operations personnel.

hardware/procedures to yield expected results, and hardware/system improvements. Requirements for non-firing type data were so great that such data were collected for most scheduled hot firing tests during a time interval between propellant loading and actual hot firing. The following are some of the special tests which were conducted:

- SSME cold gimballing
- Liquid oxygen tank pressure undershoot
- Liquid hydrogen tank vent flow rate
- Calibration of 100 percent propellant loading sensors
- Evaluation of propellant temperature stratification
- Helium bottles blow down
- Liquid oxygen/liquid hydrogen tank self pressurization

Two examples of "add on" test requirements are: (1) anti-geyser line removal and (2) operation of hydrogen recirculation pumps at increased speed–500 Hertz versus the normal 400 Hertz power frequency.

Data collected with the anti-geyser line for preventing fluid geysering provided clues that safe operation may be possible without the anti-geyser line. Line removal would save money and vehicle weight. Deliberate data evaluation/analysis and careful planning were necessary because of serious safety implications. A series of tests were planned and successfully conducted, thus proving the line could be removed. Today's Space Shuttle does not use this line.

The recirculation pumps, one per engine, circulate fluid through the engine, liquid hydrogen hardware, and feedlines prior to engine start command for purposes of thermal conditioning. Serious difficulties were experienced in early testing relative to an imbalance between heat input to the hardware and pump capacity. Doubts regarding pump flow/delta pressure adequacy existed; however, pump replacements without program impact were not possible. Increased pump speed offered increased flow quantity and pressure rise and consequently could resolve a potentially marginal condition without schedule impact. The test was conducted and system performance established for future use if necessary.

MAIN PROPULSION TEST ACCOMPLISHMENTS

This section presents a listing and discussion of: important issues representing accomplishments of the program; unworkable designs/procedures which were made workable by changes and then verified; workable designs/procedures for which adjustments to achieve acceptance were initially anticipated and were accomplished; improved hardware for performance, safety; and other issues. The listing cannot be considered complete and cannot be

compared line by line with test requirements of Table 3-1, which includes initial requirements plus requirements added during the program. Nevertheless, this specific listing qualitatively demonstrates the worthiness of propulsion system test programs for complex vehicle designs. Twenty-five percent of the listing is estimated to have resulted from "add on" test requirements during the test program. This statistic is most important and demonstrates the knowledge and discipline to prevent oversights and to enable the necessary design insight. Complex development programs without a flexible propulsion system test can expect major unpleasant technical, schedule and cost surprises.

To enhance the understanding of the accomplishments, issues with some similarity have been grouped under arbitrary titles as follows:

- Liquid Hydrogen Recirculation Pump Operational Procedures/Hardware Changes
- Liquid Oxygen Engine Pump Chill Down System and Oxygen Tank Loading
- Liquid Hydrogen Tank Loading
- Firing Initiation
- Firing Termination
- Miscellaneous

To further enhance insight, each individual issue has been related according to two other factors as indicated in parentheses after each issue title: (1) classification of the issue relative to consequences—catastrophic; unworkable; workable with adjustments, as was originally anticipated; and product improvement, with implementation being optional; and (2) time phasing of operation in which the issue occurs—preflight or flight. Table 3-2 is repeated from the main body of the report and provides a summary as to distribution of issues according to identified categories.

*Table 3-2. **Reported Propulsion System Testing Accomplishments Classified by Consequence and Time Phased — MPTA***

Stage	Catastrophe		Unworkable		Workable Mod. Expected		Improvement		Total Per Stage
	Flight	Preflight	Flight	Preflight	Flight	Preflight	Flight	Preflight	
Shuttle	3	3	5	17	3	6	1	2	40

Liquid Hydrogen Recirculation Pump Operational Procedures/Hardware Changes

High Point Bleed Sequencing/Line Back Pressure (unworkable, preflight). Hydrogen gas, created within the propellant delivery system during periods of zero or low flow rates, collects at the high point of the 17-inch feedline—immediately downstream of the ET/Orbiter disconnect. A 3/4 inch diameter line from this location, through the Orbiter/ground interface and connecting with the facility vent line, permits removal of this collecting gas. In order to conserve liquid hydrogen propellant in the system, a programmed mode of operation was adopted for the control valve within this line. The valve would open prior to recirculation pump start and close prior to tank pressurization preparatory to engine start. Unfortunately, excessive delivery system heat leak and other factors resulted in a large gas volume which the high point bleed could not remove; thus recirculation pumps would not operate (due to cavitation) and engine start was impossible. Procedures were modified to open the high point bleed line valve during propellant loading prior to formation of the large gas volume.

High back-pressure at the vent line exit also reduced line flow potential, thus relocation of the line exit was required. Incorporation of both hardware and procedural changes led to successful removal of the gas volume, successful operation of hydrogen recirculation pumps, and acceptable propellant temperatures at SSME pump inlets and throughout the feedlines. Successful operation was demonstrated on all subsequent firings of the propulsion system.

Propellant Loading Temperature (unworkable, preflight). Initial vehicle propellant loading operations used propellants from supply barges. Supplied propellants were anticipated to be approximately the same quality as supplied from storage tanks at KSC. Vehicle liquid hydrogen recirculation pumps, exposed almost directly to this supplied/incoming flow, cavitated, thus revealing the influence of incoming liquid hydrogen temperature on pump performance. A procedural change requiring propellant supply barge venting to reduce liquid hydrogen temperature several hours prior to planned vehicle propellant loading resolved the problem and provided valuable insight for necessary KSC operations.

500 Hertz Recirculation Power (workable - mod expected, preflight). It was concluded that liquid hydrogen propellant delivery system thermal conditioning (recirculation pump operation, high point bleed operation, and system heat leak) was sensitive to a number of variables and that the current design was marginally acceptable. Normal pump operation used a 400 hertz power supply; however, all involved hardware could accommodate operation with 500 hertz power. Operation at 500 hertz would increase recirculation pump flow and pressure rise capabilities, thus increasing system margin. A contingency test was performed successfully, and a performance data base was developed for future use as appropriate.

Restart Procedure (improvement, preflight). Possible events can result in liquid hydrogen recirculation pump cutoff and consequently require restart at some later time. Procedures for reinitiating recirculation pump start existed and were proven acceptable during testing; however, the procedures were unduly complex.

A procedure which is simpler and requires less time was developed and verified to be acceptable.

Liquid Oxygen Engine Pump Chilldown System and Oxygen Tank Loading

Liquid Oxygen Drain Back Time (workable - mod. expected, preflight). The major part of liquid oxygen replenish flow does not progress into the ET liquid oxygen tank, but is diverted from the orbiter liquid oxygen manifold into the three 12-inch diameter feedlines as engine pump/equipment chilldown flow. Liquid oxygen within the propellant delivery system upstream of the orbiter manifold is virtually stagnant, relatively warm, and will not satisfy engine start requirements. Prior to engine firing, the feedline liquid oxygen is exchanged by colder liquid oxygen from the ET liquid oxygen tank. This is accomplished by terminating replenish and continuing engine chill flow with all liquid oxygen flow now supplied from the ET liquid oxygen tank. Propellant conditions within the feedline and engine thermal conditions were investigated for various times of liquid oxygen drainback from the ET. The optimum time of nine minutes was selected and procedures finalized. (Drainback time was driven to five minutes by other requirements well into the flight program.) The ET liquid oxygen loading manifest accommodates propellant drainback quantities. The provided data base allows flexibility in adjusting the liquid oxygen load at KSC.

Inerting Purge Procedures (workable - mod. expected, preflight). Propellant tanks, valving, and lines require inerting prior to exposure to propellants. A continuous flow procedure is used where an inert gas is introduced into the tank simultaneously with expulsion from tankage of a similar quantity of gas. This process continues until strategically located instruments are recording safe concentration levels within tankage. Humidity levels are also evaluated. Testing to minimize time required for inerting and inert gas helium quantities required were conducted and procedures were finalized.

Facility/Vehicle Chill (workable - mod. expected, preflight). Transition of facility liquid oxygen supply lines from ambient temperature to liquid oxygen temperature is a critical phase relative to avoiding vehicle hardware damage. The fluid dynamics encountered as fluid changes state in the transfer lines can result in "slugging" which is destructive to vehicle hardware. While test site and launch site hardware differ in many respects, carefully prepared procedures are necessary for each, and experience from one site benefits the other site. For example, the ET was one of a limited number of stages loaded at the launch site which did not experience hardware damage, due to such experience from test sites.

Propellant Loading Temperature and Slow Fill (catastrophic, preflight). As discussed for liquid hydrogen earlier, liquid oxygen temperatures from supply barges were unacceptably warm. The relatively warm liquid oxygen created the potential for a geyser which could result in devastating consequences. A procedural change to vent supply barges several hours prior to vehicle loading was implemented, thus reducing the temperature of supplied propellant. Liquid oxygen geysering can also result from the transfer of stored engine thermal energy to liquid oxygen within the propellant delivery system. The engine liquid oxygen

chill flow is beneficial for removing engine thermal energy. Procedural changes were made which prevented propellant loading beyond vehicle station 1616 prior to liquid oxygen chill flow expelling propellant warmer than -281°F through the engine bleed valve. This is accomplished by restricting slow fill rate to 75 gpm until the necessary temperature is achieved.

Tank Loading Under Pressure (unworkable, preflight). This subject applies to both liquid oxygen and liquid hydrogen. External tank structural deficiencies required positive internal pressure during loading of propellant tanks. This was contrary to initial design requirements and required changes in the loading method and procedures. The method adopted was to control tank pressure by cycling vehicle vent valves, a method not previously used and inherently introducing risk. The approach was explored and thoroughly developed, and procedures were established. Liquid oxygen replenish rates to ensure proper flight levels were also accomplished at elevated tank pressures while cycling vent valves. The ET structural capabilities changed frequently necessitating changes to the procedure. Tank redesign to preclude this condition remains impractical, thus the procedures are permanent.

Closed Loop Propellant Level Control (unworkable, preflight). Space Shuttle propellant loading at KSC is automated, while at the propulsion system test site, only replenish/topping phases were planned to be automated. Extensive data collection was necessary to properly implement automation. Initial test results from the test site demonstrated that the control electronics could not satisfy requirements and that replacement of control electronics was required. Similar changes to the launch processing system (LPS) at KSC were required. Availability of replacement control electronics resulted in extensive testing to collect necessary data for total system development and verification.

Re-Initiating Replenish (unworkable, preflight). Contingency planning required procedures for replenish flow termination and subsequent re-initiation. Initial procedures for liquid oxygen resulted in a large pressure spike when the fill-and-drain valves were opened to re-initiate replenish. The cause was release of stored energy from self-pressurized, trapped liquid oxygen in the facility lines. New procedures to eliminate/reduce the pressure spike were developed and verified for use at both test and launch sites.

ET Oxygen Off Loading (catastrophic, preflight). Procedures developed for off-loading liquid oxygen from vehicle to facility resulted in unacceptable initial flow rates within Orbiter hardware and unacceptable pressure spikes in facility hardware. An orifice was installed at the Orbiter/facility interface to reduce both the transient and steady state flow rate. Procedures were modified to reduce facility surge pressure.

Anti-Geyser Line Removed (improvement, preflight). Anti-geyser line removal increased payload capability and afforded significant cost savings. Important development work using subsystem test hardware for the initial configuration led to an understanding of system performance, later verified on MPTA, and the operating margin. From this work, the idea for removing the anti-geyser line was conceived.

The concept was demonstrated on MPTA and later was accepted. Oxygen geysering is a critical safety issue, thus procedures must be carefully developed and verified for all normal and contingency operating modes.

Hydrogen Tank Loading

Topping Flow Rate (workable - mod. expected, preflight). Orbiter lines were shown by test to have higher pressure loss than predicted. An increase in interface pressure was required to achieve the ICD topping flow rate.

Tank Thermal/Tank Loading Under Pressure (unworkable, preflight). The liquid hydrogen tank, similar to the liquid oxygen tank, required internal pressure during loading and with propellant on board. Vent valve cycling was again used to satisfy the requirement. An additional requirement related tank propellant level to bulkhead temperature during initial loading phases. Techniques for accomplishing this later requirement were developed. Procedures for both requirements were developed and verified. Since launch and test site facilities are not identical, similar procedures were developed for KSC.

Firing Initiation

Engine Ready Logic Changes (unworkable, preflight). Initially "engine ready" activity completion, necessary for engine start, was accomplished immediately prior to start command, thus causing its failure to result in recycling the countdown, which was undesirable. To minimize roll back, engine ready was revised with completion planned immediately following ET prepressurization. This provided the flexibility to hold the countdown for strategizing/action without recycling. Procedures were amended and satisfactory operation was demonstrated many times.

Countdown Interlock Reduction (unworkable, preflight). Difficulty in achieving necessary conditions for a successful countdown resulted in reassessment of interlocks. The number of interlocks was reduced and many were simplified to enhance launch probability.

Aft Fuselage Inerting/Thermal Control (workable - mod. expected, preflight). Two objectives were considered in this development: (1) assuring that components, both propulsion and electrical, do not become excessively hot or cold during prefiring and firing periods; and (2) assuring that gas sampling devices are properly located to detect gas leaks (primarily preflight). Variables involved are quantity, temperature, discharge location, and direction or purge gases and gas sampling probe locations. Gas sampling systems for flight were developed. With a new vehicle design, particularly if several verification/development firings are to be made, a higher than normal purge system capability is necessary for safety. These systems have been instrumental in preventing total stage/facility loss. All above aspects were an important part of the program, and launch redlines and procedures were developed accordingly.

Feedline Screens (unworkable, preflight). Oxygen and hydrogen feedline screens located near low pressure (oxygen and hydrogen) engine pump inlets provide protection from foreign particles. Current usage on Shuttle is a result of the test program. Loss of needle bearings from the oxygen drain valve and distribution of these bearings throughout the hydrogen and oxygen systems was the motivator. Screen usage affected available engine NPSP, which was marginal; thus the design approach was critical. Valve redesign was necessary; a replacement spool piece was required to permit testing to continue.

Hydrogen Pressurant Diffuser Structural Failures (unworkable, preflight/flight). The pressurant diffuser, located internal to the liquid hydrogen tank for distributing pressurant gas throughout the tank ullage and achieving required tank pressures and minimizing pressurant gas weight, structurally failed as did several improved designs. Flow induced vibration created by incoming high velocity pressurant gas resulted in metal fatigue. Change of material from aluminum to steel resolved the issue. Component testing had not experienced this difficulty although testing was incomplete.

Oxygen Tank Prepressurization Overshoot (unworkable, flight). Liquid oxygen tank ullage pressure exceeded the allowable control band during prepressurization. The pressurizing gas is helium which is supplied from the facility. Flow capability is regulated by appropriate facility valve, and orifice and ET located pressure sensors provide necessary flow control. The problem was resolved by eliminating a short delay in the SATS program and removing the helium supply valve actuator orifices. Subsequently, ullage pressure requirement was maintained within the control band.

Pressure Transducer Drift (unworkable, preflight). Pressure transducers mounted to or in close proximity to hardware containing cryogenic fluids yielded inaccurate pressure readings. Some measurements were crucial for firing initiation and safe operations. It was necessary to relocate transducers which were used for establishing engine ready. Other transducers were removed from the control logic while others were insulated to minimize drift.

ET Pressure Transducer Failure (unworkable, flight). Flow induced vibrations, created by propellant tank vent valve cycling required to maintain a positive tank pressure for structural considerations, resulted in fatiguing the poteniometer-to-wiper interface of the pressure transducer. Transducer redesign was necessary. System environments identify problems which otherwise are not identified because testing is regulated by input requirements, which frequently are limited due to lack of system knowledge.

POGO Accumulator Gas Collapse (unworkable, preflight). An accumulator containing both liquid and gaseous oxygen located on the SSMEs between the low pressure and high pressure pumps changes response frequencies of the propulsion system. During engine start, the accumulator ullage gas, oxygen, collapsed and pressure spikes in the upstream feedline were observed. Incorporation of a helium gas precharge into the accumulator prior to engine start

resolved the problem. A helium precharge was incorporated into the normal engine start sequence.

Feed System Stability (workable - mod. expected, flight). POGO (interaction of vehicle structure with dynamically created disturbances occurring within the oxygen feedline/engine system) is a major concern for vehicle designers. Complicated math models necessary to establish vehicle stability margin prior to vehicle flight require input data which are obtained from special ground testing utilizing flight configuration hardware. Extensive data of the oxygen feed system/engine were obtained with dynamic disturbances established by applying pressure pulses of varying magnitude, frequency, and mode shape upstream of the pump inlet. This was accomplished both with and without hardware which was provided for preventing dynamic problems. Quantifying safety margins was vital data for the program.

Engine Start Positioning (improvement, preflight). Unknown magnitude of sideloads during the SSME start transient required flared outward positioning of engine nozzles to provide maximum clearance in the event the TVC relief valve relieved and the engine moved. Testing demonstrated engines could be safely started in the preferred null position. Procedures were changed to start in the null position.

Aft Heat Shield Over-Pressure (unworkable, preflight). Hydrogen gas exits SSME nozzles for a brief time period immediately prior to engine start. Ignition of hydrogen, which may accumulate in the vehicle base under some conditions, can result in unacceptable pressure spikes at engine ignition. A system to prevent accumulation by burning hydrogen as it exits SSME nozzles can prevent pressure spikes from occurring. Propulsion system testing confirmed that a burn-off system was required. A burn-off system was designed for the launch site and was verified during propulsion system testing.

ET/Orbiter Oxygen Propellant Disconnect (catastrophic, unworkable, preflight/flight). This valve is located immediately downstream of a 90 degree elbow and near the aft end of the ET. It is a 17-inch diameter quick disconnect (two valve butterfly - poppet concept). The turning flow forces generated by the close proximity of the upstream elbow could overload the valve actuating mechanism resulting in unplanned rapid valve closure which could be disastrous. Fortunately this was discovered prior to MPTA hot firing. This event, differing from most others cited herein, is an example of subsystem rather than full system testing benefits; however, the proper environment must be used to achieve representative data. Unfortunately, in many cases, system level testing is the only source of the correct environmental conditions and sometimes hardware exclusions restrict such environments during system testing—the following issue is an example.

Facility to ET Orbiter Hydrogen Umbilicals (unworkable, preflight). This is an example of hardware exclusions restricting environments for system level testing—noted immediately above. Umbilicals were not a part of the orbiter to test site interface for MPTA, and a flight configuration ground and airborne disconnect was not used for the ET. Extensive difficulties were experienced at the launch site on

both of these connections, and yet the actual launch site procedure had been performed many times at the test site.

Firing Termination

Solid Nitrogen Formation in Engine MFV (catastrophic, preflight). Main fuel valve leakage after engine cutoff was an infrequent problem. Diffusion of the nitrogen engine purge into the fuel system and resultant solidification were corrected by engine software modifications permitting the helium fuel system purge application once each hour after propellant loading to remove the nitrogen.

Liquid Oxygen Low Level Cut Off (catastrophic, flight). Ability to use the maximum quantity of onboard oxygen is supportive of maximizing vehicle payload. The liquid oxygen low level cutoff system is included to allow maximum usage of oxygen and yet prevent engine pump operation with inadequate NPSP, which can be destructive/hazardous. Propulsion system test termination by the liquid oxygen low level cutoff system was successfully demonstrated for differing test conditions—numbers of engines operating, engine thrust level, etc. Optimum time delay—time interval between cutoff sensor dry indication and engine shutdown command—was determined for various engine combinations. Extensive development activity relative to sensor performance, location, etc., was necessary prior to conducting of meaningful propulsion system testing.

Liquid Hydrogen Low Level Cutoff (workable - mod. expected, flight). Sensors for liquid hydrogen low level cutoff are located within the liquid hydrogen tank near the tank bottom. The primary purpose of the liquid hydrogen low level cutoff system is prevention of an oxygen rich shutdown and engine operation with inadequate NPSP. Hydrogen residuals are small because propellant density is low. The low level cutoff system was successfully demonstrated and engine performance was shown to be adequate.

Liquid Hydrogen Dump Sequence (workable - mod. expected, flight). Hydrogen residuals within the propellant delivery system for the orbiter must be removed for safing purposes for any normal or aborted mission. RTLS dump valves located on the orbiter near the ET/orbiter disconnect can safe the system for an aborted mission. For normal missions, hydrogen is dumped overboard through the fill-and-drain line. Tests verified proper valve sequencing and line pressurizing with helium for inerting for both the RTLS and fill-and-drain valve options. An additional venting option was evaluated and determined to be feasible.

Liquid Oxygen Prevalve Closure Demonstration (unworkable, flight). The liquid oxygen high pressure pump available NPSP for flight was predicted to be zero during last engine cutoff. Operation at zero NPSP is hazardous. A procedure to prevent zero NPSP operation, to rapidly close the prevalve and pressurize the downstream feedline was devised and demonstrated.

Miscellaneous

Nitrogen Purge Termination (unworkable, preflight). SSME nitrogen purge supply pressure decay time was established to define a sequence at KSC.

Orbiter/ET Umbilical Cavity Purge (unworkable, preflight). The required purge flow for the orbiter/ET umbilical varied significantly as a result of varying leakage rates; thus the existing "fixed" purge supply approach was inadequate. Modifications were necessary. A regulated supply concept was implemented at both the test and launch site.

Liquid Nitrogen Formation (unworkable, preflight). Liquid nitrogen exited the intertank drains, and aft fuselage cameras determined the presence and sources for liquid nitrogen in the aft fuselage. Improvements primarily to engine insulation to prevent local liquid nitrogen generation and modifications to prevent liquid nitrogen impacting sensitive hardware were made in the intertank and aft fuselage.

Freezing Hydraulic Warmant Flow (unworkable, preflight). Testing revealed that SSME MFV and CCV actuators could not be maintained within required temperature limits to prevent freezing with warmant hydraulic flow. Heaters were added to the MFV actuators and a warming purge was directed over the CCV actuators. A number of special tests of extended time duration were required to obtain appropriate data and validate the solution.

Steer Horn Failure (catastrophic, flight). A flight configuration engine nozzle line containing hydrogen failed at the steer horn—external engine line carrying hydrogen to and from nozzle coolant tubes—during hot firing. Extensive hydrogen leakage resulted, and burning external to the vehicle occurred. The potential cause of line failure was probably interaction between engines firing in a cluster configuration. Steer horn redesign was required.

Liquid Oxygen Pump Bearing Spalling (workable - mod. expected, flight). Liquid oxygen high pressure pump bearing spalling was a problem during single engine development testing. Engine liquid oxygen bleed valve cycling was a successful "fix" for the bearing spalling problem. Bleed valve cycling was demonstrated during propulsion system testing and incorporated into the launch site countdown.

MPS HARDWARE PROBLEMS BY TEST

The preceding section on accomplishments, while including some hardware issues, emphasizes system issues. Hardware performance is also a primary requirement to be verified. This section provides information on orbiter/ET and engine hardware although the two are separated. Presented data on both orbiter/ET and engine hardware, as well as the summary, Table 3-3, repeated from the main body of the text (Table 13), demonstrates a lack of hardware maturity during earlier testing. Without tests to uncover and help resolve the issues early in the program, many of these failures/experiences will occur on flight vehicles.

Orbiter/ET Hardware Problems by Test

This section provides a tabulation by test of some of the MPS hardware problems experienced during the MPTA program. The tabulation is illustrative and is not complete. Repeated problems from test to test are frequently listed only once. The following guide will help to interpret the tabulated information.

Table Interpretation Guide

- Pre SF-X and SF-X
  - Selected Major Events
  - ⇑ Prefiring

  - ⇓ Test Events
    - (x) = Firing Duration and y = Date

- Stage propulsion hardware related - non engine
- Non-stage propulsion hardware related - engine

Note: (1) Most engine GSE items not included
(2) Repetition between firings omitted in most cases
(3) SF-X = Static Firing No. X

Table 3-3. Hardware Replacement and Repair by Test for MPTA

MPTA Test Number	Engine	Pump	Controller	Nozzle	Major Valves	EIU/MOM	Other: Flowmeter/gimbal bearing/POGO/Preburner/L.O. Seal	Control Instrumentation	Prevalve	17" Disconnect	Level Sensors	Fill and Drain Valve	LH2 Recirculation System, Pumps, High Point Bleed Valves	Pressurization Orifices Valves, Sensors	Auxiliary Drain Valve	Hazardous Gas Detection - Fire Detection	Cut-Off Sensors	LH2 Diffuser	Control Instrumentation	Other: Feedline Screens, Loading Orifice, Drain Orifice
	←			ENGINE				→	←					VEHICLE						→
Resonant 40% O2																				
Resonant 40% O2																				
Loading Test																				
1-001																				
1-002							1		3	2			4			4				1 Screen
2											1									2 Screen
3							1					line					1	1		1 GO2 Vent
4																		1	1	
5-A	1	12			9						4							1		2 Vents
5						1				1		line		4		1	2		1	
6-01				1	9	1													2	
6-02/3		1			7		1	1					3			4	1	1		
6-04	1		1	3		1										2	2			
7-01					1															
7-02					2									2			4			
8					2					1				5					1	
9-01		1									2						1		1	
9-02		4				1			1					1			1		1	
10				3	4	10					2			1						
11-01		2			7				6					4			2			
11-02				3					3	1				6						
12				3								1								
Total	2	20	1	13	41	15	3	1	13	5	9	3	7	23	0	11	13	4	7	6

Note: Hardware changes made prior to designated test number

NDH001

- Pre SF 1 and SF 1
  - Prevalve closure rate slowed
  - 17" O_2 disconnect flapper removed
  - Redesigned AFT heat shield for increased loads

  - Engine 2 LH_2 prevalve position indicator erroneous
  - LO_2 level sensor failed to indicate
    - (1.5 sec) 04/21/78

- Pre SF 2 and SF 2
  - O_2 vent orifice installed. Load under pressure. ET structural problem

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  - Many HFA, LFA and acoustic measurement failures
  - LH_2 tank diffuser failed
  - MFV 3 leak
  - ET/ORB attach fittings over-torqued
    - (20 sec) 05/18/78

- Pre SF 3 and SF 3

  - Two auto sequence holds - LH_2 recirculation valve failed to close
  - Engine 1 O_2 prevalve position switch failed
  - LH_2 level sensor failed
    - (42 sec) 06/15/78

- Pre SF 4 and SF 4
  - ET wall corrosion (under insulation) - discovered post test during 9 months down period
    - (104 sec) 07/07/78

- Pre SF 5A and SF 5A
  - Installed liners in LO_2 F&D disconnect
  - Installed new 17" O_2 disconnect
  - Installed flight design feedline screens
  - Changed torque on aft ET/orbiter fittings
  - Installed remote mounted engine inlet pressure transducers
  - Installed steel GH_2 ET Diffuser
  - O_2 tank minimum pressure of 1.7 PSIG instituted

  - Main fuel valve leak
  - LO_2 and LH_2 100 percent sensors 3 and 4 failed
  - EIU intermittent problems
  - Recirculation pump electrical and mechanical problems
  - EIU problem. Two changeouts
    - (1.5 sec) 05/04/79

- Pre SF 5 and SF 5

  - LO_2 ullage pressure offset 2 PSI
  - Full duration test aborted by faulty pump radial accelerometer
  - Numerous EIU problems
  - Various EIU and MDM problems
    - (54 sec) 06/12/79

Note: (1) Test 6-01; 19 sec of planned duration on 07/02/79 - engine problems resulting in major stage repairs
(2) Test 6-02; 0 sec of planned duration on 10/24/79 - Faulty H_2 gas controller within aft compartment
(3) Test 6-03; 10 sec of planned duration on 11/04/79 - engine steer horn failure

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- Pre SF 6-04 and SF 6-04
  - O_2 engine cutoff sensors moved from orbiter to ET
  - Installed He tank vent line venturi

  - Lost several temperature/pressure red line measurements
  - H_2 concentration excessive in aft compartment
  - EIU swap for trouble shooting
    - (555 sec) 12/17/79

- Pre SF 7-02 and SF 7-02
  - Outboard LO_2 fill-and-drain valve mechanism failure - changeout
  - One LH_2 prevalve changeout
  - Six pressurization orifices enlarged
  - Baffle added to both 100 percent LO_2 sensors (liquid level)
  - Two step turbine discharge temperature redline incorporated for start
  - Changed LO_2 bleed flow rate/temperature/time required
  - Increased engine stagger time - reduce overpressure

  - Faulty closed indicator on LO_2 auxiliary drain valve
  - LO_2 ECO sensors responded slowly
    - (555 sec) 02/28/80

- Pre SF-8 and SF-8
  - All GH_2 pressurization orifices enlarged

  - LO_2 engine cutoff sensors indicated wet until after cutoff
  - Several orbiter flight transducers failed
    - (541 seconds) 03/20/80

- Pre SF 9-02 and SF 9-02
  - Removed baffle from 100 percent LO_2 sensor 2
  - EIU replaced
  - LH_2 flight type ullage transducers installed

  - 17" interconnect valve failed to seat
    - (574 sec) 05/30/80

- Pre SF 10 and SF 10
  - Position indicator switches on 17" disconnect replaced
  - Capacitance probe installed for O_2 engine cutoff
  - One EIU replaced
  - Minimum LH_2 tank pressure requirement of 3 PSI during replenish

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  - Engine fuel preburner burn through aborted test
    - (106 sec) 07/12/80

- Pre SF 11-02 and SF 11-02
  - New design O_2/H_2 ullage pressure transducers
  - Maintain 2 PSIG minimum in O_2 tank for levels above 20 percent
  - Repaired 17" LH_2 disconnect valve

  - LH_2 auxiliary drain valve would not close after detanking
    - (588 sec) 12/04/80

- Pre SF 12
    - (624 sec) 01/17/81

Engine Hardware Problems by Test

This section provides a tabulation by test of the engine (component) hardware problems experienced during the MPTA program. The following guide will help to interpret the tabulated information.

Table Interpretation Guide

- Pre SF-X and SF-X
  - Selected Major Events
  - ⇑ Prefiring

  - ⇓ Test Events
    - (x) = Firing Duration and y = Date

Note: (1) Most vehicle experiences not included
(2) GSE experiences not included
(3) SF-X = Static Firing No. X

- Pre SF-2 and SF-2
  - No gimballing
  - Start to 70 percent thrust

  - MFV leak after firing
  - LO2 flow meter straighteners cracked
    - (20 sec) 05/18/78

- Pre SF-3 and SF-3
  - Replaced leaking MFV
  - No gimballing
  - Start to 70 percent thrust. Maximum = 90 percent
    - (104 sec) 07/07/78

- Pre SF-4 and SF-4
  - No gimbaling
  - Thrust limited to 90 percent

  - Engine controller channel A electronics failed
  - All O2 flow meters removed post test
    - (104 sec) 07/07/78

- Pre-SF-5A and SF-5A
  - One engine changeout
  - Several (each) H2 and O2 pump changeout - both high/low pressure
  - Flight nozzles replace stub nozzles
  - MOVs modified to prevent buzz problem
  - No gimballing - LPOP ball strut galling

  - MFV leak - burning vehicle base area
  - Nozzle replaced - bad steer horn weld
    - (1.5 sec) 05/04/79

- Pre SF-5 and SF-5
  - Changed turbopump feedline bolts
  - Gimballing limited by LPOP strut galling

  - Controller channel B, one engine required post firing replacement
  - Pump accelerometer aborted test .
  - POGO suppressor bubble collapsed
    - (54 sec) 06/12/79

- Pre SF-6-01 and SF-6-01
  - Strain gages added to steer horn

  - Controller failed before firing - test rescheduled
  - MFV cracked - bad damage to vehicle and engine controller
  - POGO accumulator, one engine, failed to charge
  - One engine controller channel A failed
  - Critical flow size (fracture mechanics) became issue
  - High time components became issue
    - (0 sec) 06/39/79, (19 sec) 07/02/79

- Pre SF-6-03 and SF-6-03
  - Vehicle rebuilt after SF-6-01 problems
  - Same engines as SF-6-01 but approximately 1/2 controllers, pumps, etc., exchanged

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  - Engine turbine seal failed - test aborted
  - Steer horn ruptured at shutdown
    - (10 sec) 11/04/78

- Pre SF-6-04 and SF-6-04
  - Replaced one engine
  - Stub nozzles replaced flight nozzles
  - Replaced three turbopumps
  - Seventy percent minimum operating thrust

  - Liftoff seal for H2 pump failed to close
    - (555 sec) 12/17/79

- Pre SF-7-01 and SF-7-01

  - O2 pump turbine discharge temperature spike - aborted test
    - (5 sec) 02/01/80

- Pre SF-7-02 and SF-7-02
  - Turbine temperature redline changed - two step during start
  - O2 engine bleed flow rate change

  - H2 pump liftoff seal leaked
  - Engine anomalous start up
    - (555 sec) 02/28/80

Pre SF-8 and SF-8
  - OPOV reset
  - Seventy percent minimum operating thrust

  - H2 pump seal broken
  - Controller channel B failed
    - (54 sec) 03/20/80

- Pre SF-9-01 and SF-9-01
  - H2 pump changeout - one engine
  - Controller changeout - one engine

  - H2 pump manifold collapsed - test aborted
  - O2 pump secondary seal failed
    - (6 sec) 04/18/80

- Pre SF-9-02 and SF-9-02
  - One fuel pump replaced
  - All high pressure O2 pumps replaced - FMOF capability
  - FASCOS added - one engine
  - POGO accumulator rake removed
  - Bifurcation heat exchanger inspected for faulty material

  - Controller channel B drifted
  - Chamber pressure over-shot during transition to mainstage
  - O2 heat exchanger interface temperature excessive
    - (574 sec) 05/30/80

- Pre SF-10 and SF-10
  - Stub nozzles replaced with flight nozzle - all engines

  - Fuel preburner burn-through
  - All three engine controllers failed
    - (106 sec) 07/12/80

- Pre SF-11-01 and SF-11-01
  - Replace two O2 turbopumps
  - Changed two H2 turbopumps

  - Hole developed in engine nozzle
    - (20 sec) 11/03/80

- Pre SF-11-02 and SF-11-02
  - Flight nozzles replaced with stub nozzles

  (586 sec) 12/04/80

- Pre SF-12 and SF-12

    - (624 sec) 01/17/81

APPENDIX 4

SATURN V S-IC MAIN PROPULSION SYSTEM

The S-IC stage was the first stage/booster of the Saturn V vehicle (Figure 4-1). Other stages were S-II, the second stage discussed in Appendix 5, and S-IVB, the third stage discussed in Appendix 6.

The S-IC stage utilized the propellant combination of RP-1 fuel and liquid oxygen in its five F-1 engines. The engines were arranged in a cluster with one engine mounted in the center and the other four mounted outboard around the center engine. Thrust vector control was accomplished by gimballing the four outboard engines.Figures 4-2 through 4-6 support the stage and systems description which follows.

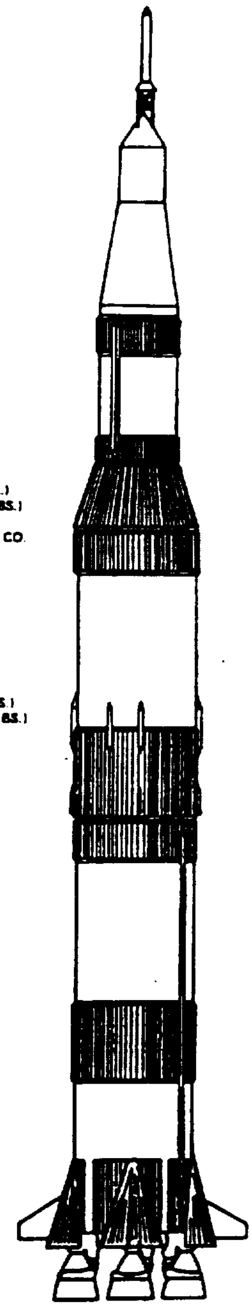
Major components of the S-IC included the thrust structure, fuel tank, intertank, liquid oxygen tank, and forward skirt. The thrust structure absorbed the punishment of the five engines and redistributed the forces into uniform loading around the base of the rocket. The thrust structure also provided support for the engines and engine accessories and miscellaneous equipment. There were four holddown posts to hold the vehicle in place prior to liftoff. Four fins mounted at the base of S-IC provided necessary stability margins for the total Saturn V vehicle.

The propellant tanks included special fill-and-drain hardware to fill the tanks at high flow rates (2000 GPM for RP-1 fuel and 10,000 GPM for liquid oxygen). The original design of the propellant system included a fuel bubbling system using nitrogen to prevent temperature stratification of the fuel. During propulsion system testing, this was determined to be unnecessary and was eliminated. Liquid oxygen bubbling with helium to prevent geysering and stratification was continued throughout the program.

Pressurization of the RP-1 fuel tank in flight was accomplished with helium drawn from four elongated aluminum bottles mounted in the liquid oxygen tank. The cold helium was heated by ducting it through heat exchangers on the F-1 engines. The liquid oxygen tank was pressurized with oxygen obtained by tapping off liquid oxygen from the engine liquid oxygen domes and ducting it through heat exchangers on the engines.

The hydraulic system used to gimbal the four outboard engines featured a somewhat unconventional approach in that it used RP-1 fuel as the actuating fluid. The RP-1 fuel was taken directly from the high-pressure fuel duct, routed to the gimbal system, then returned to the engine fuel system. To compensate for the shortcomings of RP-1 fuel as the fluid, special care was taken in the design of valves.

Brief descriptions of some of the major subsystems of the S-IC stage are contained in the following paragraphs:



LAUNCH ESCAPE SYSTEM

COMMAND MODULE

SERVICE MODULE

LUNAR MODULE

INSTRUMENT UNIT (IU)

DIAMETER: 6.6 METERS (21.7 FEET)
HEIGHT: 0.9 METERS (3 FEET)
WEIGHT: 2,040 KILOGRAMS (4,500 POUNDS)
CONTRACTOR: IBM CORPORATION

THIRD STAGE (S-IVB)

DIAMETER: 6.6 METERS (21.7 FEET)
HEIGHT: 18.1 METERS (59.3 FEET)
WEIGHT: 121,000 kg. FUELED (266,000 LBS.)
11,300 kg. DRY (24,900 LBS.)
ENGINE: ONE J-2
PROPELLANTS: LIQUID OXYGEN 89,000 kg. (196,000 LBS.)
LIQUID HYDROGEN 19,900 kg. (43,750 LBS.)
THRUST: 926,367 NEWTONS (208,242 LBS.)
CONTRACTOR: McDONNELL DOUGLAS ASTRONAUTICS CO.

SECOND STAGE (S-II)

DIAMETER: 10.1 METERS (33 FEET)
HEIGHT: 24.8 METERS (81.5 FEET)
WEIGHT: 483,318 kg. FUELED (1,087,580 LBS.)
36,478 kg. DRY (80,420 LBS.)
ENGINES: FIVE J-2
PROPELLANTS: LIQUID OXYGEN 384,000 kg. (845,713 LBS.)
LIQUID HYDROGEN 73,000 kg. (160,464 LBS.)
THRUST: 5,131,968 NEWTONS (1,153,712 LBS.)
CONTRACTOR: ROCKWELL INTERNATIONAL

FIRST STAGE (S-IC)

DIAMETER: 10.1 METERS (33 FEET)
HEIGHT: 42.1 METERS (138 FEET)
WEIGHT: 2,246,540 kg. FUELED (4,952,775 LBS.)
130,441 kg DRY (287,574 LBS.)
ENGINES: FIVE F-1
PROPELLANTS: LIQUID OXYGEN 1,471,427 kg.
(3,243,942 LBS.) RP-1 KEROSENE
642,177 kg. (1,415,257 LBS.)
THRUST: 34,096,110 NEWTONS (7,665,111 LBS.)
AT LIFT-OFF.
CONTRACTOR: THE BOEING COMPANY

FIGURE 4-1. SATURN V LAUNCH VEHICLE

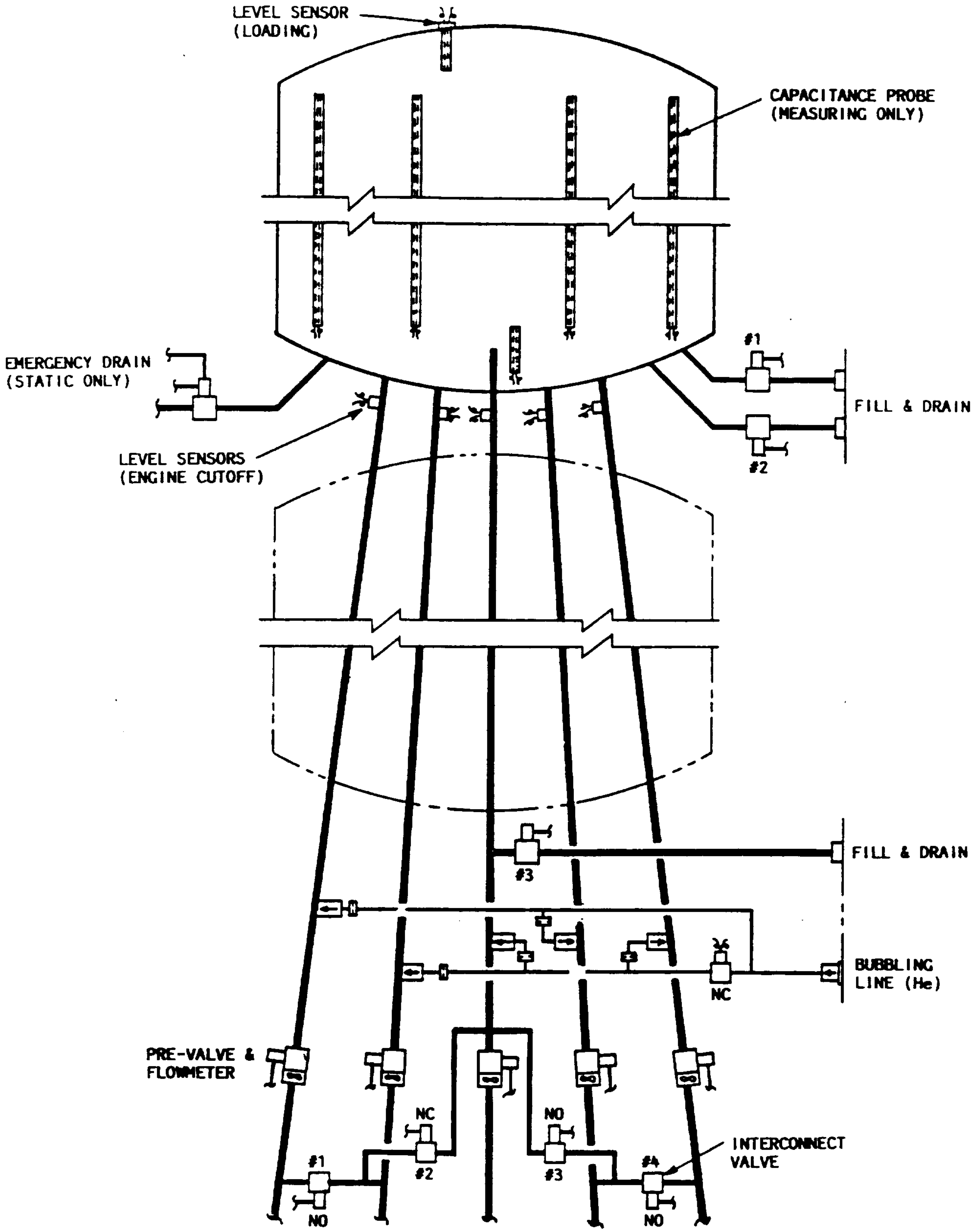


FIGURE 4-2. S-IC LOX SYSTEM

Liquid Oxygen System (Figure 4-2)

- One tank - forward position.
- Five suction lines (one per engine) routed through fuel tank tunnels.

Liquid Oxygen FILL

- Prevalves, vent valves and inter-connect valves 1, 3, and 4 in open position.
- Open liquid oxygen fill-and-drain valves 1 and 2.
- Fill rate: ~10,000 gpm (~5,000 gpm per valve).
- Mass loaded controlled by liquid oxygen loading level sensor.
- Fill and drain valve 3 available for use, if required.

Liquid Oxygen Replenish

- Required continuously until start of prepressurization.
- Replenishing controlled by liquid oxygen loading level sensor.

Normal Liquid Oxygen Drain

- Vent helium bottles and close vent valves.
- Pre-valves and interconnect valves 1, 3, and 4 in open position.
- Pressurize liquid oxygen tank (from ground)
- Open fill-and-drain valves 1 and 2, which allowed drainage of cylindrical portion of tank.
- To drain lower bulkhead and partially drain suction lines, opened liquid oxygen interconnect valve 2 and liquid oxygen fill-and-drain valve 3.

Emergency Liquid Oxygen Drain

- S-IC had double-piston, pneumatically-actuated 17-inch emergency drain valve installed for static-removed after static.

Prevalves

- S-IC used normally-open, pneumatically-closed, 17-inch valves, with redundant closing provisions using ground supplied nitrogen for static test and/or on-pad abort.

Liquid Oxygen Conditioning

- Geysering suppression by thermal pumping.
    - Suction lines interconnected to establish flow path.
    - Two separate thermal pumping systems isolated by interconnect valve 2.
    - Thermal pumping initiated during initial filling by injecting helium into suction lines 1 and 3.

- When any prevalve or interconnect valve (other than 3) was closed, helium bubbling was required in all suction lines.
- Prevalves relieved at ~50 psig.

- Pump inlet temperature controlled by thermal pumping. Initiated helium bubbling in lines 1 and 3 at T-10 minutes and continued until start of liquid oxygen pressurization. Interconnect valves closed immediately prior to ignition.

Engine Cutoff Sensor

- Inboard: liquid level sensor
  - Signal from sensor initiated time, expiration of which signaled IECO.
- Outboard: Four liquid level sensors - signals required from two of four to initiate timer, expiration of which signaled OECO.

P-U Slosh Measuring

- Five continuous capacitance probes provided data for continuous profile for determination of sloshing and consumption as function of flight time.
- No active P-U system in flight: "tailored tanking" used to insure simultaneous depletion of propellants.

Liquid Oxygen Pressurization System (Figure 4-3)

Pre-Ignition: Ground Supplied Helium

- Closed vent and relief valves.
- Helium admitted through coupling and check valve to tank.
- Liquid oxygen prepressurization switch (actuated 26.0 psia, deactuated 24.2 psia, 501 and 502; 503 and subs actuated 26.5 psia, deactuated 24.2 psia) signaled ground valve to close.
- Liquid oxygen relief switch (actuated 29.0 psia, deactuated 27.5 psia, 501 and 502; 503 and subs actuated 30.0 psia, deactuated 28.0 psia) signaled liquid oxygen vent and relief valve to open.
- Helium supply remained connected until liftoff.

Flight

- Liquid oxygen tapped off liquid oxygen dome to engine heat exchanger. Liquid oxygen transformed to gaseous oxygen and routed to common manifold through gaseous oxygen flow control valve to tank.
- Gaseous oxygen flow control valve: Designed to maintain liquid oxygen tank ullage pressure at 20.5 +/- 2.5 psia. Used liquid oxygen tank pressure as reference.
- Modulated flow to tank between an optimized 30 - 50 #/sec, with maximum flow capacity of 70 #/sec.
- Minimum flow through heat exchanger ensured by using mechanical stop (adjustable) in gaseous oxygen flow control valve.

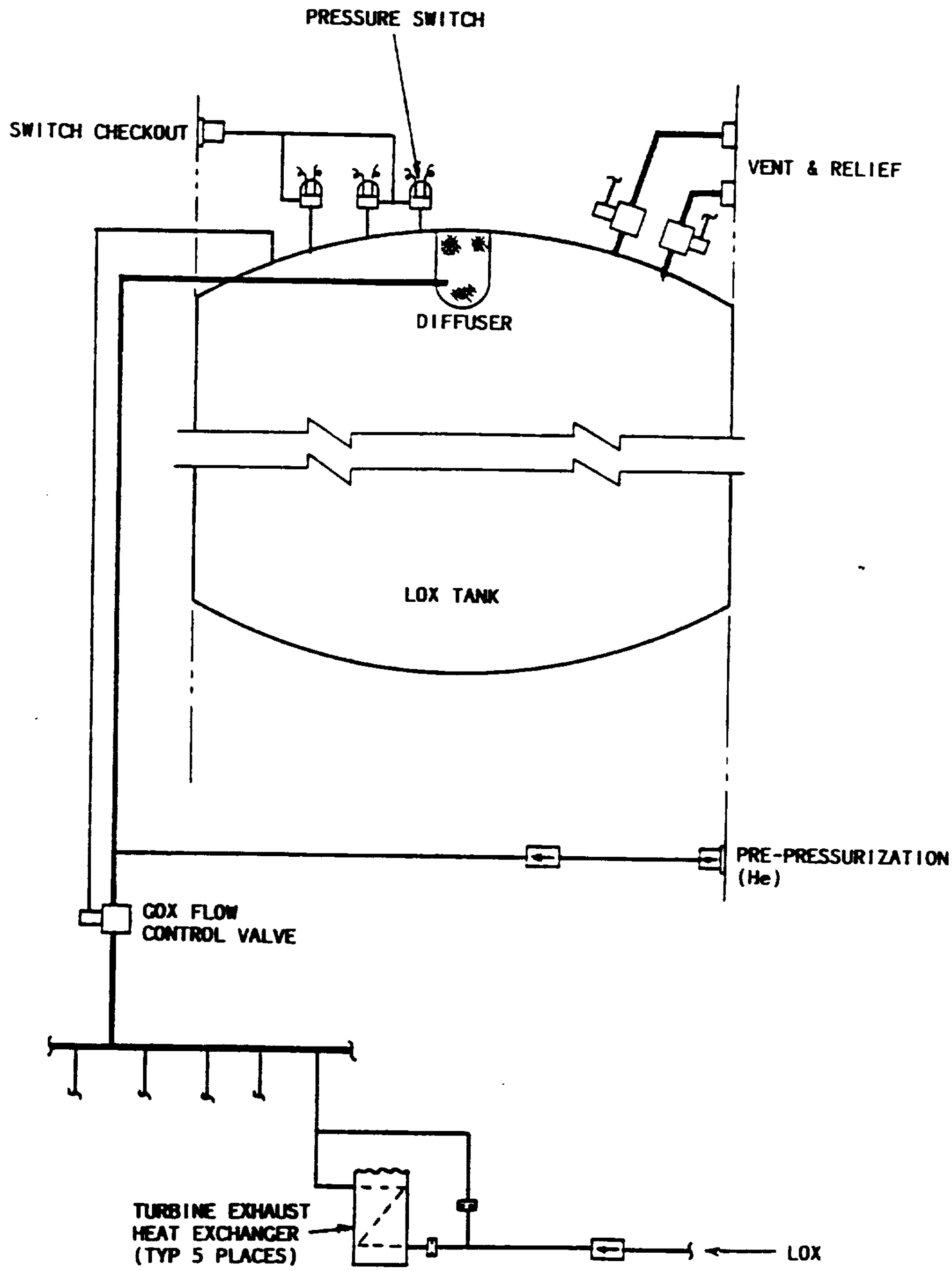


FIGURE 4-3. S-IC LOX PRESSURIZATION SYSTEM

- Liquid oxygen vent and relief valve controlled by either the liquid oxygen relief switch or the liquid oxygen vent and relief switch (actuated 25.5 psig, deactuated 23.7 psig, 501 and 502; 503 and subs actuated 25.0 psig, deactuated 22.0 psig) between ~T+65 sec and ~T+75 sec. After ~T+75 sec, the vent and relief switch assumed primary control of the valve.
- Vent and relief valve mechanically relieved at 24.0 - 25.5 psig.

Fuel System (Figure 4-4)

- One tank (aft position) RP-1 fuel
- Ten suction lines (two per engine)

Fuel Fill

- Prevalves and vent and relief valve(s) in open position
- Opened N.C fill-and-drain valve (6" diameter)
- Fill rate: ~2000 gpm
- Mass loaded controlled by liquid oxygen loading level sensor
- Temperature sensors (total of nine, three sets of three) in tank for density determination

Normal Fuel Drain

- Opened vent-and-relief valve
- Opened fill-and-drain valve
- Drained by gravity or closed vent-and-relief valve(s)
- Pressurized tank
- Opened fill-and-drain valve

Emergency Fuel Drain

- S-IC had double-piston, pneumatically-actuated 12-inch diameter emergency drain valve installed for static test— removed after static test.

Prevalves

- S-IC used normally-open, pneumatically-closed 12-inch valves, with redundant closing provisions using ground supplied nitrogen for static tests and/or on-pad abort.

Engine Cutoff Sensors

- Engine cutoff normally initiated by liquid oxygen-orientated system.
- Fuel-oriented back-up cutoff system provided (to guarantee fuel-rich cutoff).
  - Fuel bi-level sensor with redundant sensing elements at both levels— fuel passed upper level, inboard engine cutoff (IECO) initiated; fuel passed lower level, timer initiated, expiration of which signaled outboard engine cutoff (OECO).

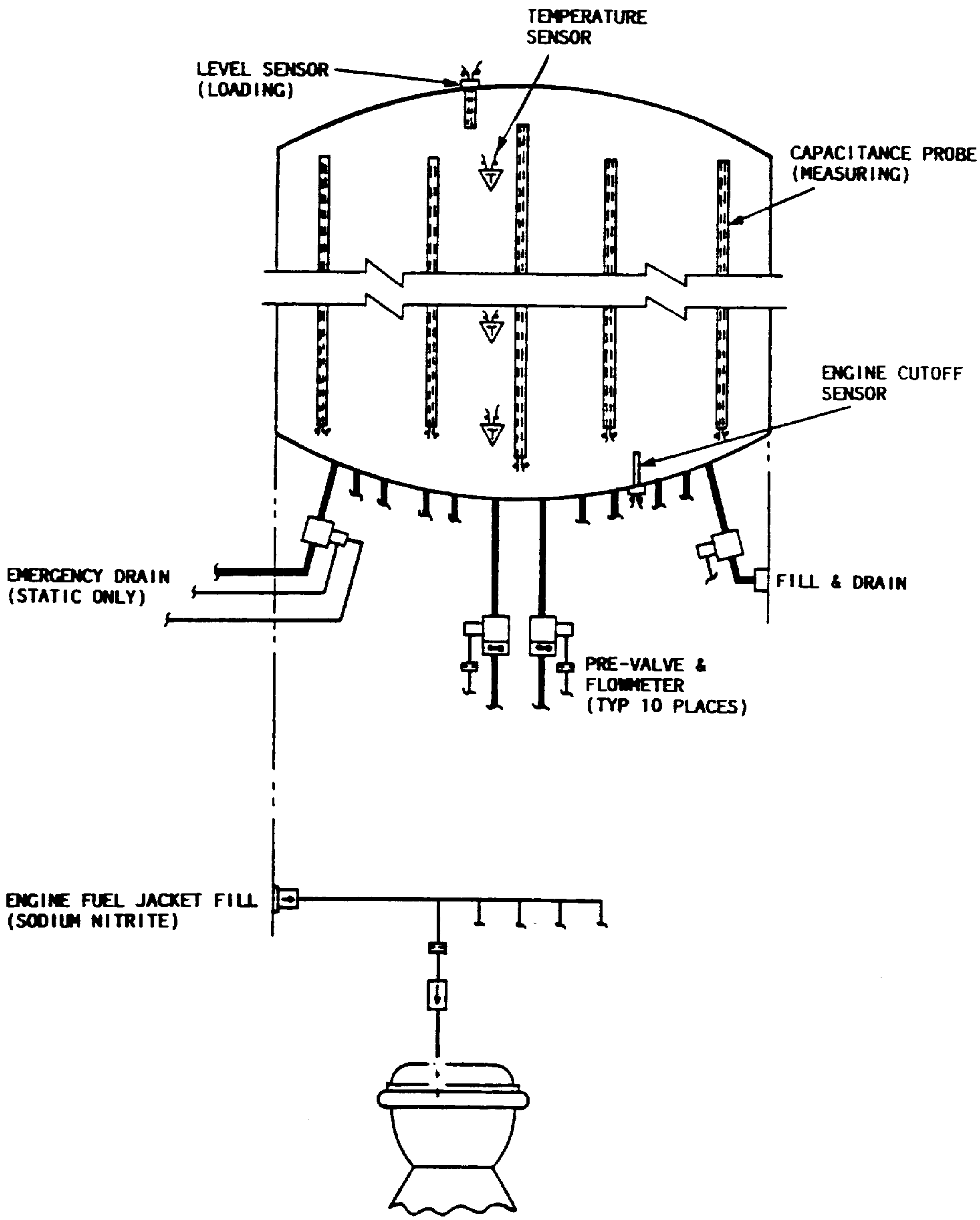


FIGURE 4-4. S-IC FUEL SYSTEM

- Cutoff circuitry for outboard and inboard completely separate. Each system armed ~5 seconds prior to predicted IECO and OECO, respectively.

Propellant Utilization (P-U) and Slosh Measuring

- Five continuous capacitance probes provided data for continuous profile for determination of sloshing and consumption as function of flight time.
- No active P-U system n flight; "tailored tanking" used to ensure simultaneous depletion of propellants.

Thrust Chamber Fuel Jacket Prefill

- Thrust chamber jacket tubes and fuel manifold were filled with sodium nitrate solution prior to engine start to aid in the prevention of rough combustion.
- Solution supplied to vehicle manifold and routed to each engine fuel manifold.

Fuel Pressurization System (Figure 4-5)

Pre-Ignition: Ground Supplied Helium

- Closed vent and relief valve(s).
- Helium admitted through coupling and heat exchangers to common manifold to tank.
- Fuel prepressurization switch (actuated 29.0 psia, deactuated 27.5 psia) signaled ground valve to close.
- Helium supply connected until liftoff.
- All five pressurization valves remained closed until liftoff.

Flight: Chilled Helium—Heat Exchanger

- Four helium storage bottles in liquid oxygen tank.
  - Pressurized to 1500 psig prior to liquid oxygen loading.
  - Pressurized to ~3150 psig after liquid oxygen loading; hi pressure OK pressure switch (actuated 3150 psia +/- 40 psia, deactuated 3010 psia minimum) interlocked in firing command circuitry. Helium temperature -283°F maximum.
  - N.C. emergency dump valve provided for rapid evacuation of bottles, if required.
- Five pressurization valves: one N. O. and four N.C.
  - Four programmed
  - One operated off pressurization switch (actuated 26.0 psia, deactuated 24.2).
- One N.C vent-and-relief valve operated off vent switch (actuated 31.5 psia, deactuated 29.7 psia). Valve mechanically relieved at 30.5 - 35.0 psia.

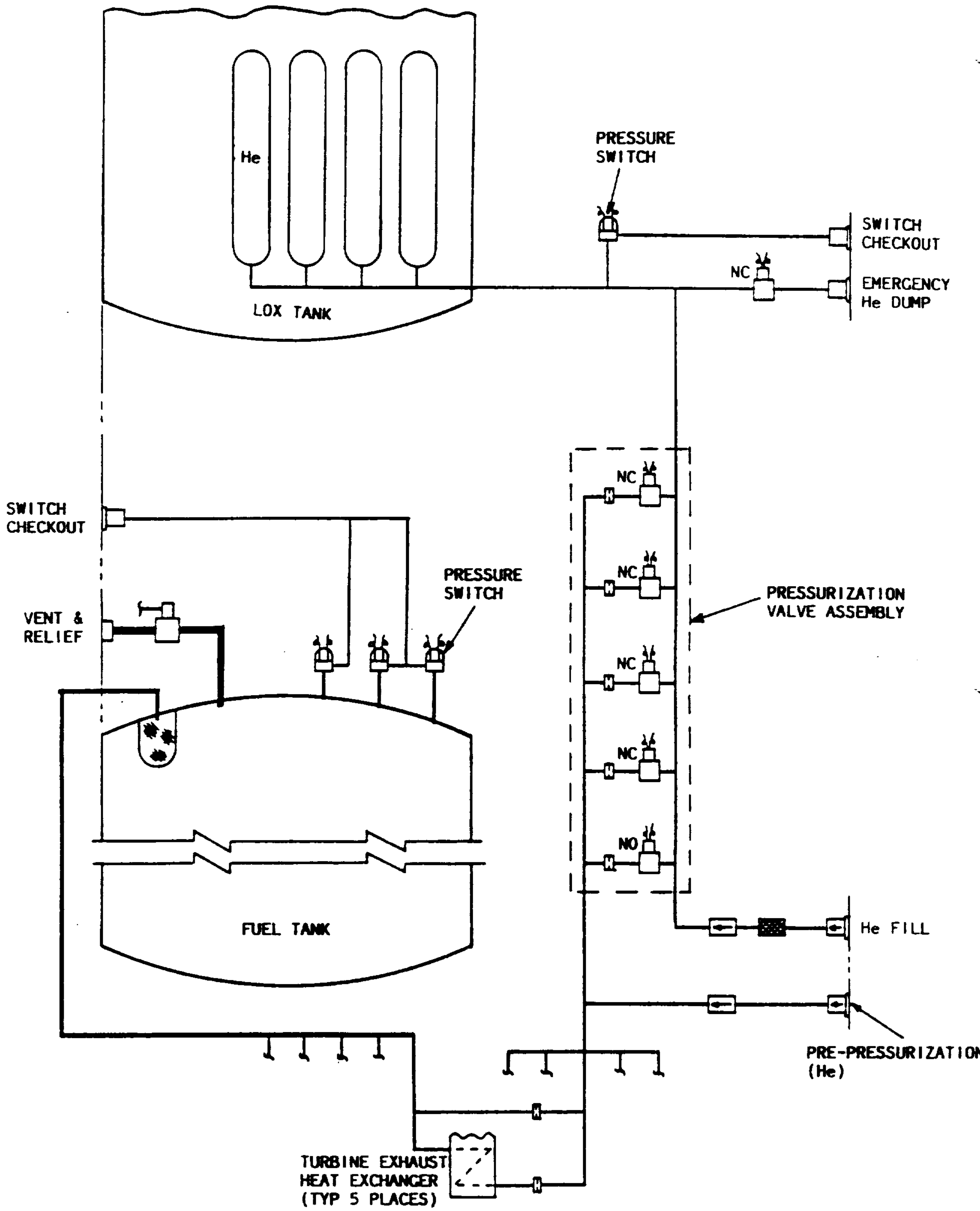


FIGURE 4-5. S-IC FUEL PRESSURIZATION SYSTEM

F-1 Engine (Figure 4-6). The general features of the F-1 engine were:

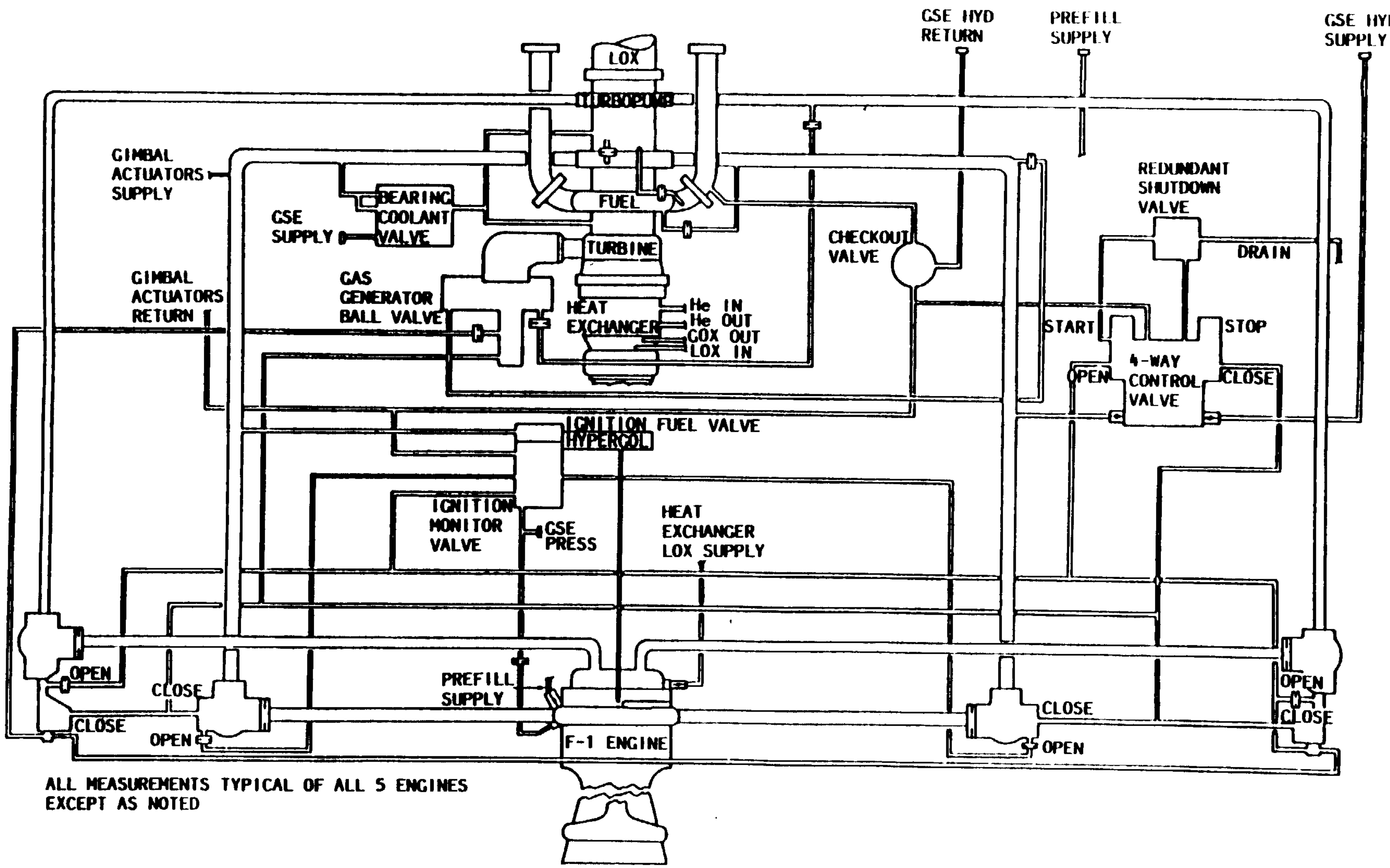
- Single start
- Fixed thrust: 1,500,000 pounds at sea level
- Propellants: liquid oxygen and RP-1 fuel
- Engine mixture ratio: 2.2:1
- Lubricant: RP-1 fuel
- Control fluid: RJ-1 for standby and start, RP-1 fuel for operation

Engine Start Sequence

- The ignition sequencer controlled start of all five engines.
- Checkout valve moved to engine return position
- Electrical signal fired igniters (four each engine)
  - Gas generator combustor and turbine exhaust igniters burned igniter links to trigger electrical signal to start solenoid of four-way control valve.
  - Igniters burned approximately six seconds.
- Start solenoid of four-way control valve directed GSE hydraulic pressure to main oxidizer valves
- Main oxidizer valves allowed liquid oxygen to flow to thrust chamber and GSE hydraulic pressure to flow through sequencer valve to open gas generator ball valve.
- Propellants, under tank pressure, flowed into gas generator combustor.
- Propellants were ignited by flame of igniters.
- Combustion gas passed through turbopump, heat exchanger, exhaust manifold and nozzle extension.
- Fuel rich turbine combustion gas was ignited by flame from igniters.
  - Ignition of this gas prevented backfiring and burping.
  - This relatively cool gas (approximately 550° C) was the coolant for the nozzle extension.
- Combustion gas accelerated the turbopump, causing the pump discharge pressure to increase.
- As fuel pressure increased to approximately 26,400 grams per square centimeter (375 psig), it ruptured the hypergol cartridge.
- The hypergolic fluid and fuel were forced into the thrust chamber where they mixed with the liquid oxygen to cause ignition.

Transition to Mainstage

- Ignition caused the combustion zone pressure to increase.
- As pressure reached 1,400 grams per square centimeter (20 psig), the ignition monitor valve directed fluid pressure to the main fuel valves.
- Fluid pressure opened main fuel valve.
- Fuel entered thrust chamber. As pressure increased, the transition to mainstage was accomplished.
- The thrust OK pressure switch (which sensed fuel injection pressure) picked up at approximately 74,500 grams per square centimeter (1060 psi) and provided a THRUST OK signal to the IU.



ALL MEASUREMENTS TYPICAL OF ALL 5 ENGINES EXCEPT AS NOTED

FIGURE 4-6. F-1 ENGINE SCHEMATIC

Engine Cutoff Sequence

- Cutoff initiated by electrical signal which energized the Stop Solenoid (part of the four-way solenoid valve).
- Actuation of Stop Solenoid closed the pressurization port, venting entrapped fluid and directing closing pressure to the GG valves, MOVs and MFVs.
- MOVs closed before MFVs to provide fuel-rich cutoff.

Engine Gimbal System

- Incorporated on outboard engines.
- The gimbal system positioned the outboard engines by gimballing the entire engine with two servo valve and actuator assemblies, which used high pressure fluid (RJ-1 from GSE during standby and start, RP-1 fuel from fuel pump outlet during engine operation) as the actuating medium.
- First gimbal system to use engine fluid at engine turbopump pressure to gimbal engine for stage thrust vector control.
- Operational pressure: 1800 - 2200 psi
- Rated flow at rated load: 120 gpm
- Rated load: 72,000 pounds
- Rated velocity: 5 deg/sec minimum
- Engine gimbal angle: +/- 5 deg

S-IC PROPULSION SYSTEM TEST OBJECTIVES

The major objective of the S-IC propulsion system test program was the establishment of the feasibility of making a major step upward in thrust from 1.5 million pounds for S-IB to 7.5 million pounds thrust for S-IC.

Additional objectives were:

- Establish functional compatibility between the ground support equipment and the stage systems
- Establish adequacy of operating procedures
- Evaluate performance of the thrust vector control system
- Evaluate fuel and oxidizer feed systems performance
- Evaluate fuel and oxidizer onboard pressurization systems
- Evaluate propellant loading systems and procedures
- Evaluate liquid oxygen geyser prevention system and procedures
- Evaluate performance of flight instrumentation and telemetry system

- Evaluate structural capability of the stage
- Evaluate optimum engine start-up sequence
- Verify F-1 engine performance in a clustered configuration

A total of 15 development static firings were required to satisfy these objectives of which three were full duration.

S-IC TEST ACCOMPLISHMENTS

Following is a listing of accomplishments achieved by the S-IC propulsion system test program. Each issue has been classified as to consequence and time phasing for which the issue may apply as was done in Appendix 3 for Space Shuttle.

Hydraulic Supply Line Failure (catastrophic, flight). The S-IC stage was the first to use RP-1 fuel as the hydraulic fluid in the stage hydraulic system. During static test S-IC-13 on the all systems test stage, the flight supply line from the fuel high pressure duct to the gimbal filter manifold on engine position 2 failed structurally and leaked. This presented a fire hazard, as well as the possibility of impairing the performance of the stage TVC system. A redesign of the supply line was made to prevent a recurrence of this problem.

Inadequate Liquid Oxygen Pump Seal Purge Supply (catastrophic, flight). The F-1 engine liquid oxygen pump seal formed a barrier which prevented liquid oxygen from the liquid oxygen pump and the RP-1 from the fuel pump from coming in contact with each other. The cavity in which the seal was located was purged with nitrogen during operation to carry any seal leakage overboard. Mixing of the liquid oxygen and RP-1 would create a hazardous situation. The original S-IC design incorporated only one 1.27 cubic foot storage sphere of nitrogen. During the propulsion system test program, it was determined that this volume was inadequate for the planned S-IC flight duration. A redesign added two additional storage spheres for nitrogen.

Loose Helium Bottle Retainer Nuts in Liquid Oxygen Tank (catastrophic, flight). The S-IC stage used helium to pressurize the liquid oxygen tank during flight. As a weight saving measure, the helium storage bottles were located inside the liquid oxygen tank. Results from the propulsion system test program revealed that the retainer nuts holding the storage bottles in place were becoming loosened during propulsion system operation. A redesign was made to fasten the nuts more securely.

Failure of Engine Mount Bolts (catastrophic, flight). During a post-test inspection following a propulsion system static firing, it was determined that three 1 3/8 inch bolts which attached the center engine to the stage thrust structure had failed. Investigation found that the bolt material was not suitable for use in these mount bolts. A redesign changing material for the bolts was made.

Terminal Countdown Sequencer Deficiency (unworkable, preflight). The terminal countdown sequencer (TCDS) was discovered to have a design deficiency which caused it to output all its commands at one time should a malfunction or loss of power supply occur. Since this occurrence might cause a catastrophic failure, the TCDS was modified for future use.

Deletion of Fuel Bubbling System (improvement, preflight). The original design of the S-IC stage incorporated a fuel bubbling system to prevent temperature stratification in the fuel tank. Data obtained during the propulsion system test program proved that temperature stratification in the fuel tank was not a problem and that fuel bubbling was not necessary. Fuel bubbling was deleted from the flight stages resulting in simplification of procedures and hardware and weight reduction.

Liquid Oxygen Pump Seal Purge Orifice Removal (improvement, flight). The initial S-IC design incorporated stage mounted orifices in the liquid oxygen pump seal purge supply lines. Data from the propulsion system tests showed that the orifices were unnecessarily restricting the purge flow rate below the specification requirement. Removing the orifices resulted in the required flow rate being obtained and simplified the stage design.

Liquid Oxygen Low Level Cutoff System Calibration (workable, flight). For flight stages which use liquid oxygen for an oxidizer, a low level liquid oxygen cutoff system is necessary to obviate the possibility of a liquid oxygen rich cutoff occurring with possible catastrophic results. During the S-IC propulsion system test program, liquid oxygen low level cutoff sensors were installed. These sensors were monitored during tests in order to gather data on their operation for the purpose of properly calibrating them for flight stage use.

Erroneous "Dry" Signals From Liquid Oxygen Depletion Sensors (unworkable, flight). During the S-IC propulsion system test program, it was found that bubbles in the liquid oxygen engine feedlines could rise past the liquid oxygen low level cutoff sensors giving an erroneous "dry" signal which could trigger engine cutoff. This required that provisions be made for a propellant system reset command to be initiated from the stage switch selector. This was necessary to prevent launch delays because of erroneous cutoffs during launch countdown.

Ice and Frost Formation on Engine Injector Faces (unworkable, preflight). Pretest checkout for tests in the S-IC propulsion system test program revealed that ice and frost were being formed on the injector face of the F-1 engines. Ice and frost on the injectors could be deleterious by plugging holes in the injector and affecting the engine mixture ratio. Investigation revealed that the following three conditions were required for injector frosting to occur:

- Liquid oxygen admitted to the engine for sufficient time to "cold soak" the liquid oxygen dome and injector
- Thrust chamber prefilled

- Liquid oxygen dome service purge in use

Based on these findings, two procedure adjustments were made:

- The thrust chamber prefill operation was accomplished prior to liquid oxygen tanking.

- Use of the liquid oxygen dome service purge was minimized when liquid oxygen was on board.

Thrust Chamber Inert Pre-Fill GSE Malfunction (unworkable, preflight). As a means to reduce the likelihood of rough combustion in the F-1 engine, the thrust chamber tubes were pre-filled with an inert liquid which acted as a fuel lead in the combustion chamber and cushioned the engine against a hard start. The original design of the GSE which was to be used for the pre-filling operation incorporated full sensors which were to terminate the filling operation automatically when the proper level was reached. Experience with the equipment during the test program showed that the automatic fill feature did not function properly because solution sloshing during flow activated the full sensors prematurely. Based on this experience, the automatic filling control circuits were deleted.

Erroneous Igniter Failure Cutoffs (unworkable, flight). The F-1 engine used igniters in the ignition sequence to initiate combustion in the gas generator and the turbine exhaust gas. Burning of the igniters was monitored by electrically recording when a link broke during the burning phase of the igniters. Originally, the igniter link break was incorporated in the engine start sequence and automatic cutoff was received if the links did not break in a timely manner. Test experience showed that the igniter lips did not always exhibit a clean break when burned and could cause an erroneous igniter failure cutoff. Circuit changes were made to remove igniter link break from the automatic cutoff circuit to prevent erroneous cutoffs.

Distorted Gimballing Drive Signals (unworkable, flight). Data obtained during the S-IC TVC system gimballing tests showed that the TVC servo amplifiers provided noisy, distorted drive signals, causing erratic positioning of the engines. A new gimbal actuator signal amplifier circuit was incorporated in the operational amplifier.

APPENDIX 5

SATURN V S-II MAIN PROPULSION SYSTEM

The S-II stage, the second stage of the Saturn V launch vehicle, was a cylindrical booster 81.5 feet tall and 33 feet in diameter. It was powered by five liquid propellant J-2 rocket engines which developed a nominal vacuum thrust of 200,000 pounds each and were ignited at altitude approximately 2 1/2 minutes after liftoff. Four J-2 engines were equally spaced on a 17.5 foot diameter circle and were mounted on gimbal bearings to provide thrust vector control while the fifth engine was mounted on the stage centerline and was fixed. A cut away drawing is shown in Figure 5-1.

The S-II stage propulsion system consisted of the propellant tanks, (included in the airframe structure), propellant systems, pressurization systems, and J-2 engines.

The propellant tanks included the liquid hydrogen tank and the liquid oxygen tank. The tanks provided structural support between the forward and aft skirts.

The liquid oxygen tank (Figure 5-2) consisted of ellipsoidal fore and aft bulkheads with waffle-stiffened gore segments. The tank was fitted with three ring-type slosh baffles to control propellant sloshing and minimize surface disturbances, and cruciform baffles to prevent the generation of vortices at the tank outlet ducts. A six port sump assembly located at the lowest point of the tank provided a fill-and-drain opening and openings for the five engine feedlines.

The liquid hydrogen tank (Figure 5-3) consisted of a long cylinder with a concave modified ellipsoidal bulkhead forward and a convex modified ellipsoidal bulkhead aft. The aft bulkhead was common to the liquid oxygen tank.

All exposed surfaces of the liquid hydrogen tank were covered with foam insulation to prevent air liquification and reduce temperature rise during cryogenic operation.

Propellant Systems

The propellant systems consisted of the fill, feed, recirculation, and propellant management systems.

The liquid hydrogen fill-and-drain system consisted of the fill valve and the airborne part of the fill disconnect coupling, while the liquid oxygen fill-and-drain system included the fill valve mounted to the liquid oxygen pump interface, 16 feet of 8-inch line with a pressure-compensating bellows, one free bellows and two universal bellows ball-joints to provide movement, and the airborne part of the fill disconnect coupling.

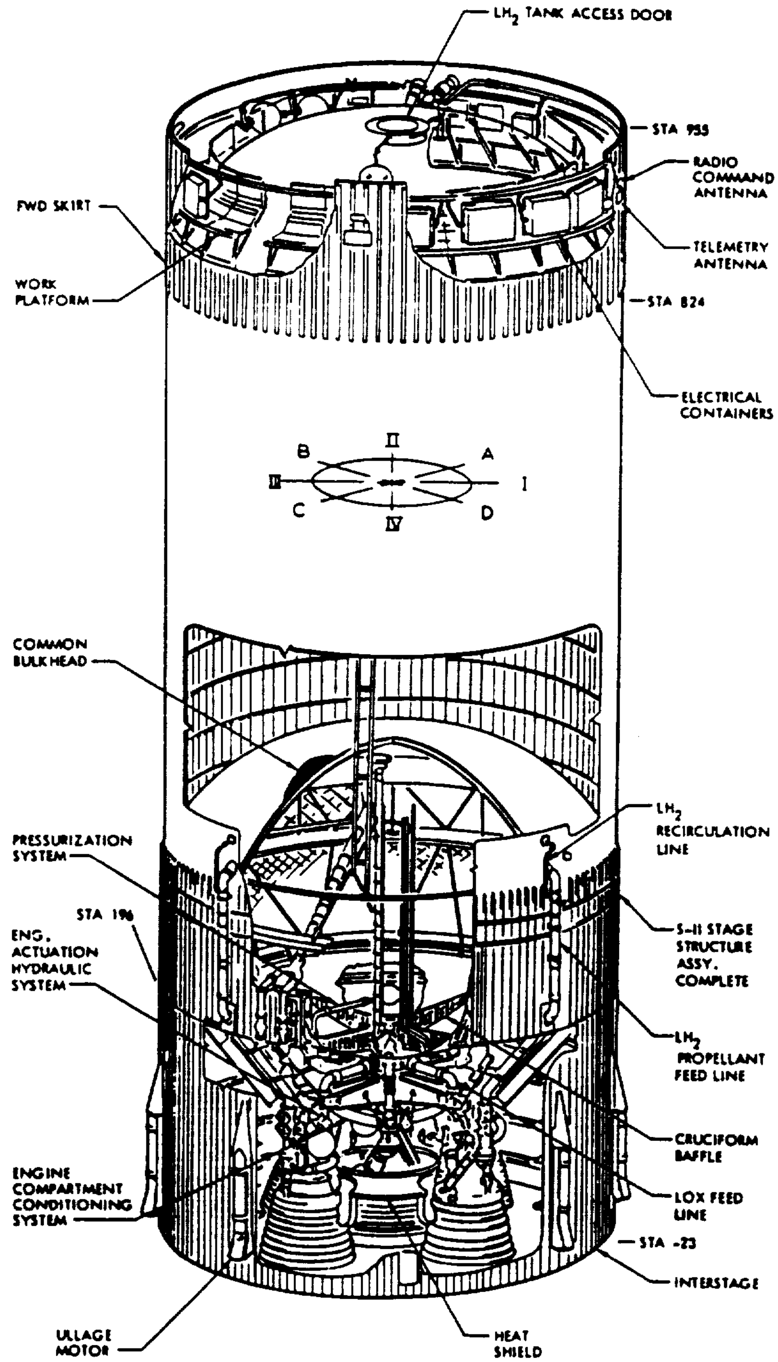


FIGURE 5-1. S-II STAGE CUTAWAY

Propellant conditioning was required to cool the vehicle feedlines and the engines and ensure required propellant temperatures were available to the engines prior to and at ignition. This was necessary to ensure acceptable engine start.

Liquid oxygen conditioning was accomplished by recirculating liquid oxygen from the tank through feed ducts and the prevalves to the engine turbopump, into the return lines and back into the liquid oxygen tank. Liquid oxygen flow was initiated after start of liquid oxygen fill and continued until T-30 minutes. Pumps were not used to recirculate flow as is often the case—flow was dependent on thermally-inducted flow. At that time, helium was injected into the liquid oxygen recirculation return line to boost recirculation and was continued until just before S-II ignition.

Liquid hydrogen recirculation was initiated at T-30 minutes and was terminated immediately prior to S-II ignition. Forced recirculation during launch consisted of closing the liquid hydrogen prevalves and operating the liquid hydrogen recirculation pumps.

The liquid oxygen feed system (Figure 5-2) supplied liquid oxygen to the five engines. The system included four 8-inch vacuum jacketed feed ducts, one 8-inch uninsulated feed duct, and five normally open prevalves. At approximately 300 milliseconds after the main valves closed, the liquid oxygen prevalves were closed. This provided a redundant shutoff for the liquid oxygen feed system.

The liquid hydrogen feed system (Figure 5-3) supplied liquid hydrogen to the five engines. The system included five 8-inch vacuum jacketed feed ducts and five normally open prevalves. The prevalves were closed following tank loading, recirculation system operation was initiated and continued until 2.2 seconds before S-II engine start at which time prevalves were opened and engines started. Approximately 425 milliseconds after the engine main valves closed, the liquid hydrogen prevalves were closed. This provided a redundant shutoff for the liquid hydrogen feed system.

Tank Pressurization

After completion of propellant loading and shortly before launch, the oxygen propellant tank was pressurized to the required level by helium from separate ground supplies. Liquid oxygen tank pressurization (Figure 5-4) from vehicle sources was initiated at S-II ignition and continued until engine cutoff. Pressurization was accomplished by using gaseous oxygen obtained by diverting a portion of the liquid oxygen supplied to the engine into the heat exchanger. The exiting gaseous oxygen flowed into a common pressurization duct through the tank pressurization regulator, and into the tank through the gas distributor. The pressurant flow rate was varied according to tank ullage pressure which was sensed by the reference pressure line from the tank.

The liquid hydrogen tank was initially pressurized with ground-supplied helium immediately prior to liftoff, as was done for oxygen. Liquid hydrogen tank

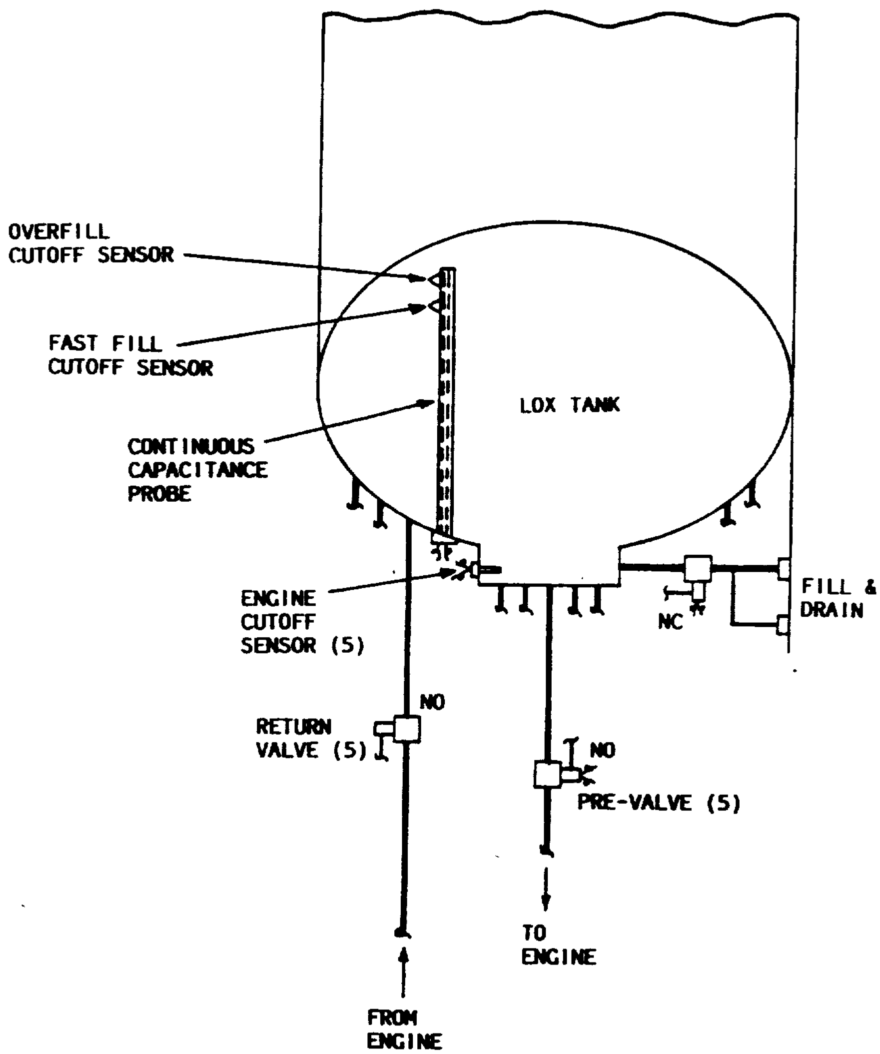


FIGURE 5-2. S-II STAGE LOX SYSTEM

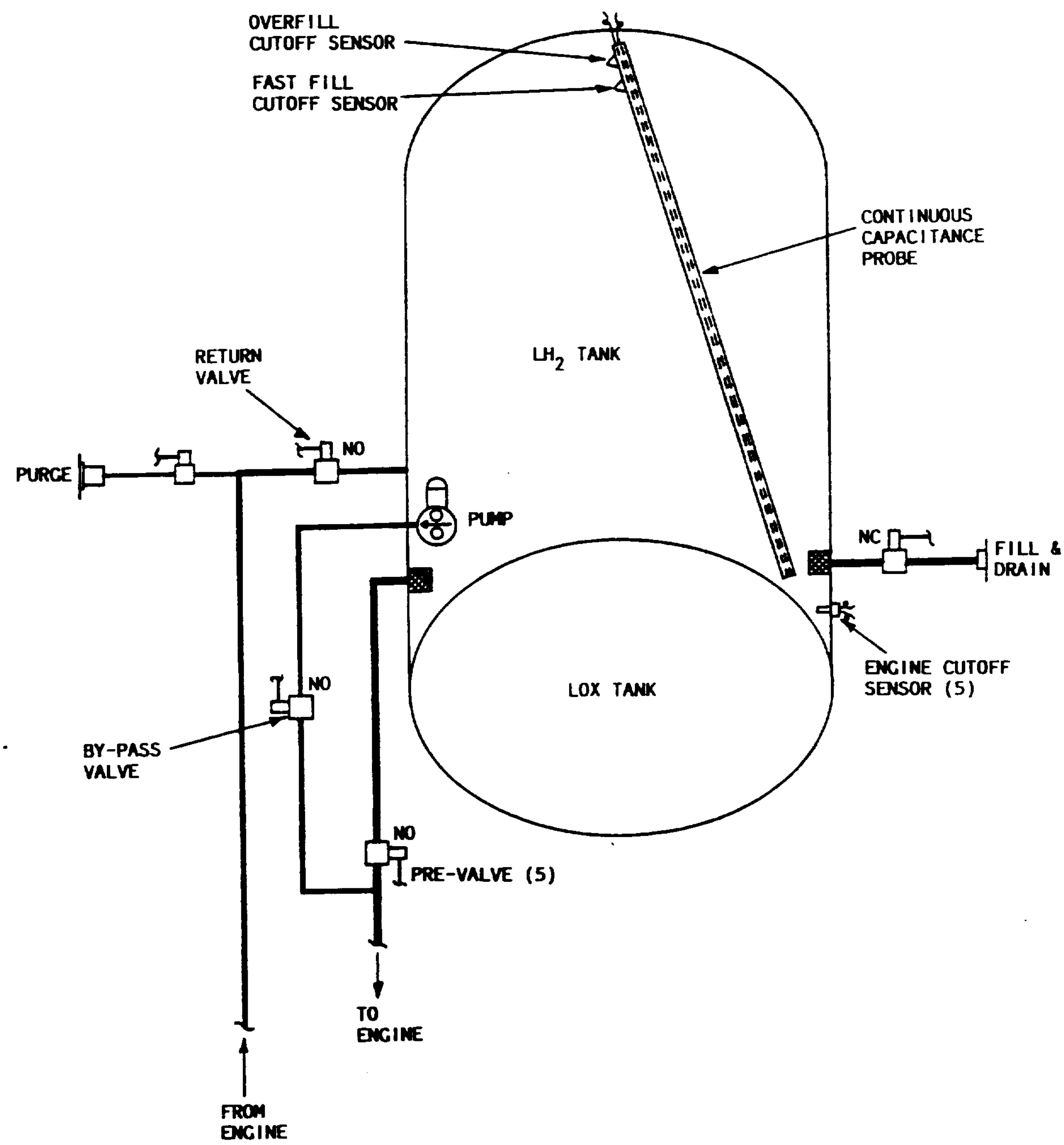


FIGURE 5-3. S-II STAGE FUEL SYSTEM

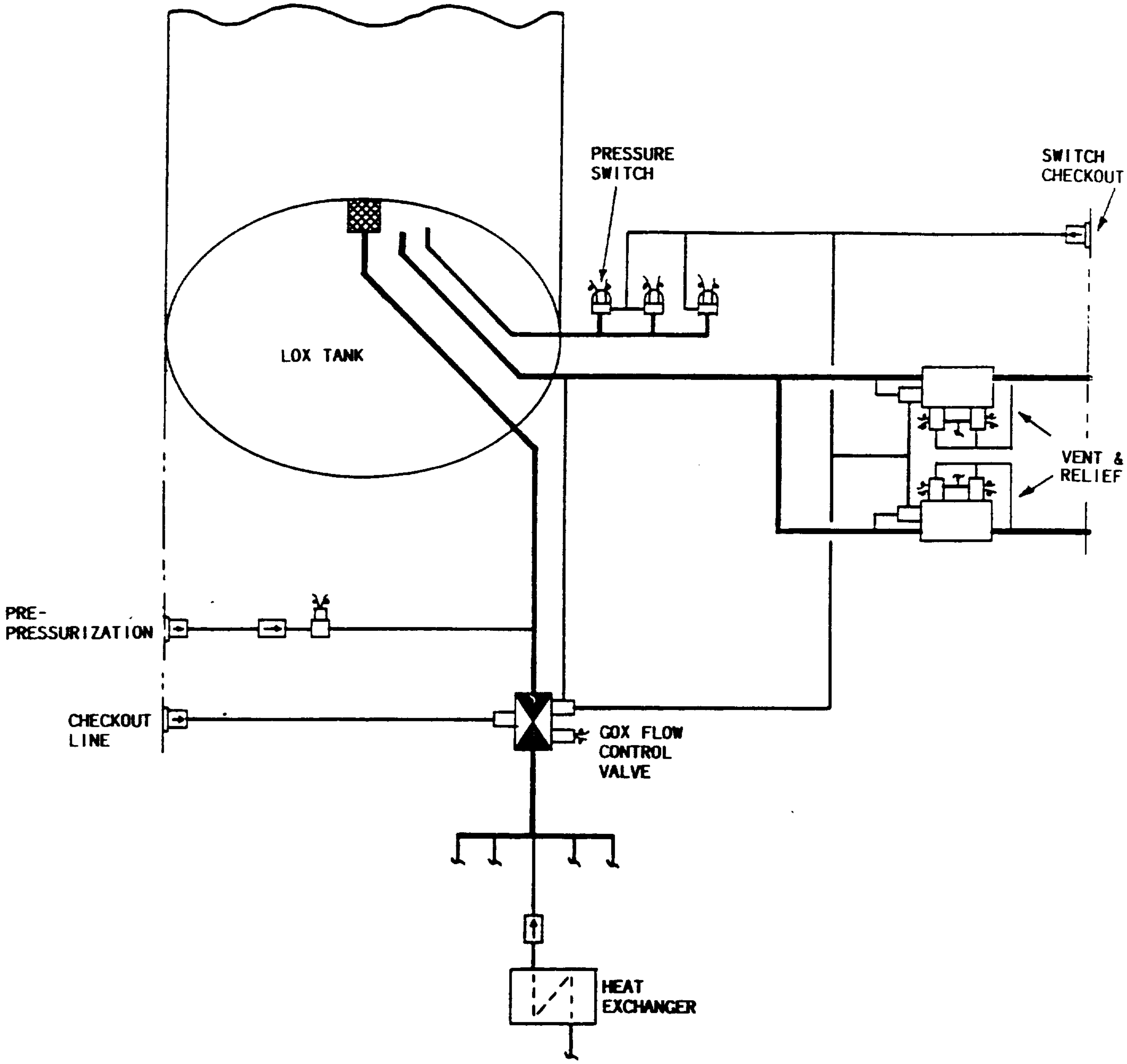


FIGURE 5-4. S-II STAGE LOX PRESSURIZATION SYSTEM

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pressurization (Figure 5-5) from vehicle sources was initiated after engine ignition. Hydrogen was bled from the thrust chamber hydrogen injector manifold of each of the four outboard engines. The hydrogen from each injector manifold flowed into a stage manifold, through the pressurization line and tank pressurization regulator (orifice for later vehicles), and into the liquid hydrogen tank ullage space via the gas distributor. The flow rate was varied according to the liquid hydrogen tank ullage pressure which was sensed by the reference pressure line from the tank.

J-2 Engine

The J-2 rocket engine (Figure 5-6) utilized the propellant combination of liquid oxygen and liquid hydrogen propellants and provided a nominal thrust of 200,000 pounds. The basic engine envelope was 116 inches long with a diameter of 80 inches. Dry weight was 3,500 pounds. Each engine incorporated self contained provisions for propulsion, control, and initial starting energy.

Primary engine propulsion components consisted of an axial flow liquid hydrogen pump, a centrifugal liquid oxygen pump, a single gas generator combustor, a tubular wall hydrogen cooled thrust chamber, a concentric orifice sweat-cooled injection plate, and a tubular wall hydrogen-cooled nozzle..

The two independently operated turbopumps were driven in series by the exhaust of the gas generator combustor and provided high pressure propellants to the thrust chamber/injector assembly. The gas generator was powered by propellants tapped off downstream of the turbopumps.

Each engine provided gas for pressurization of the stage propellant tanks. Hydrogen was tapped off downstream of the thrust chamber cooling jacket. liquid oxygen was provided by a heat exchanger mounted in the gas generator exhaust.

The S-II engine start sequence was as follows:

- Start command (occurred at T3 +1.4) engine ready.
- Augmented spark igniter (ASI) spark plugs and gas generator (GG) spark plugs fired.
- Bleed valves stopped return flow to propellant tanks.
- Oxidizer dome and gas generator oxidizer injector were purged.
- Main fuel valve allowed liquid hydrogen to flow into engine thrust chamber and into ASI.
- ASI oxidizer valve allowed liquid oxygen to flow to ASI.
- Sparks ignited the propellants in the ASI.
- Stage supplied mainstage enable signal.

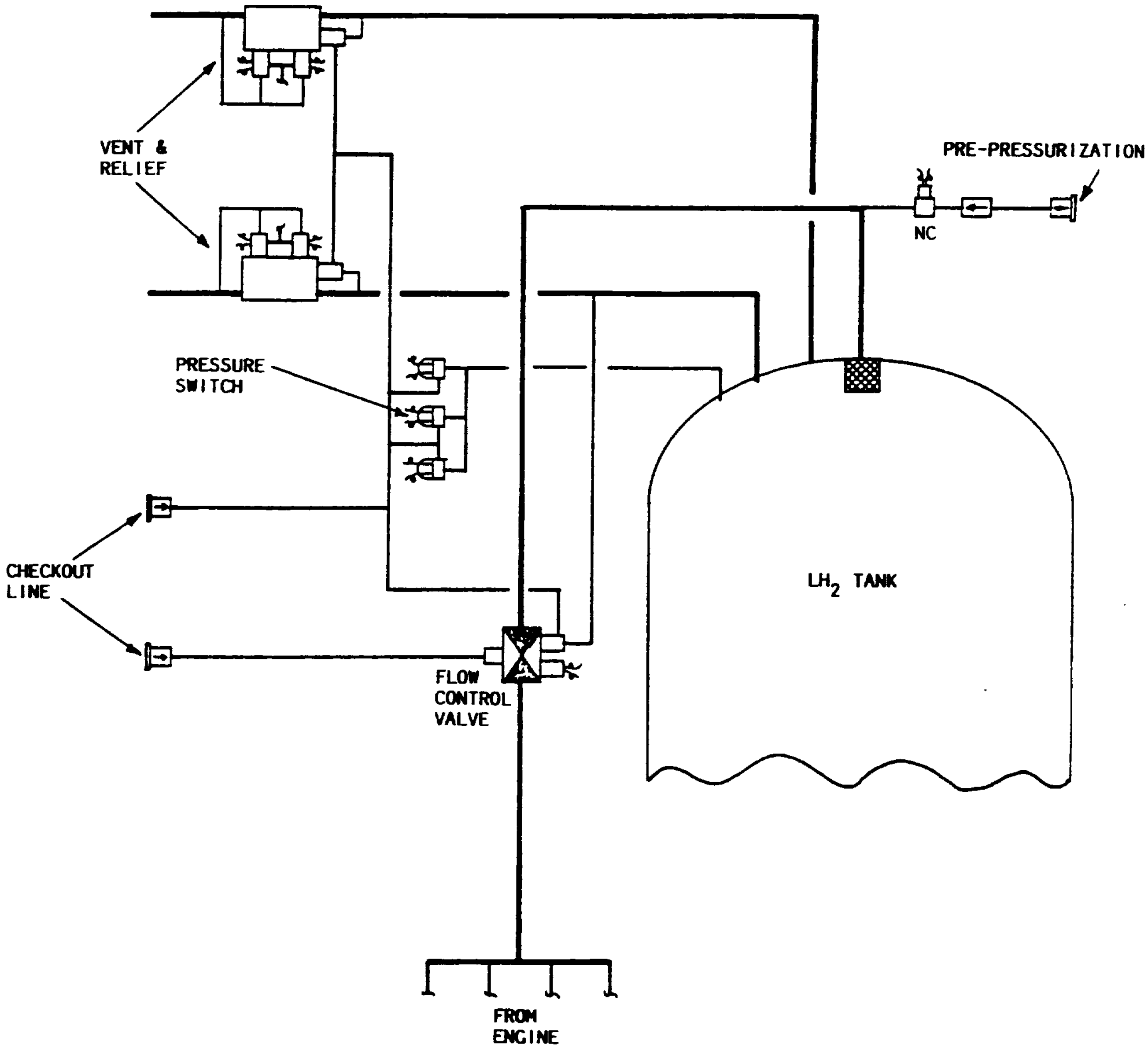


FIGURE 5-5. S-II STAGE FUEL PRESSURIZATION SYSTEM

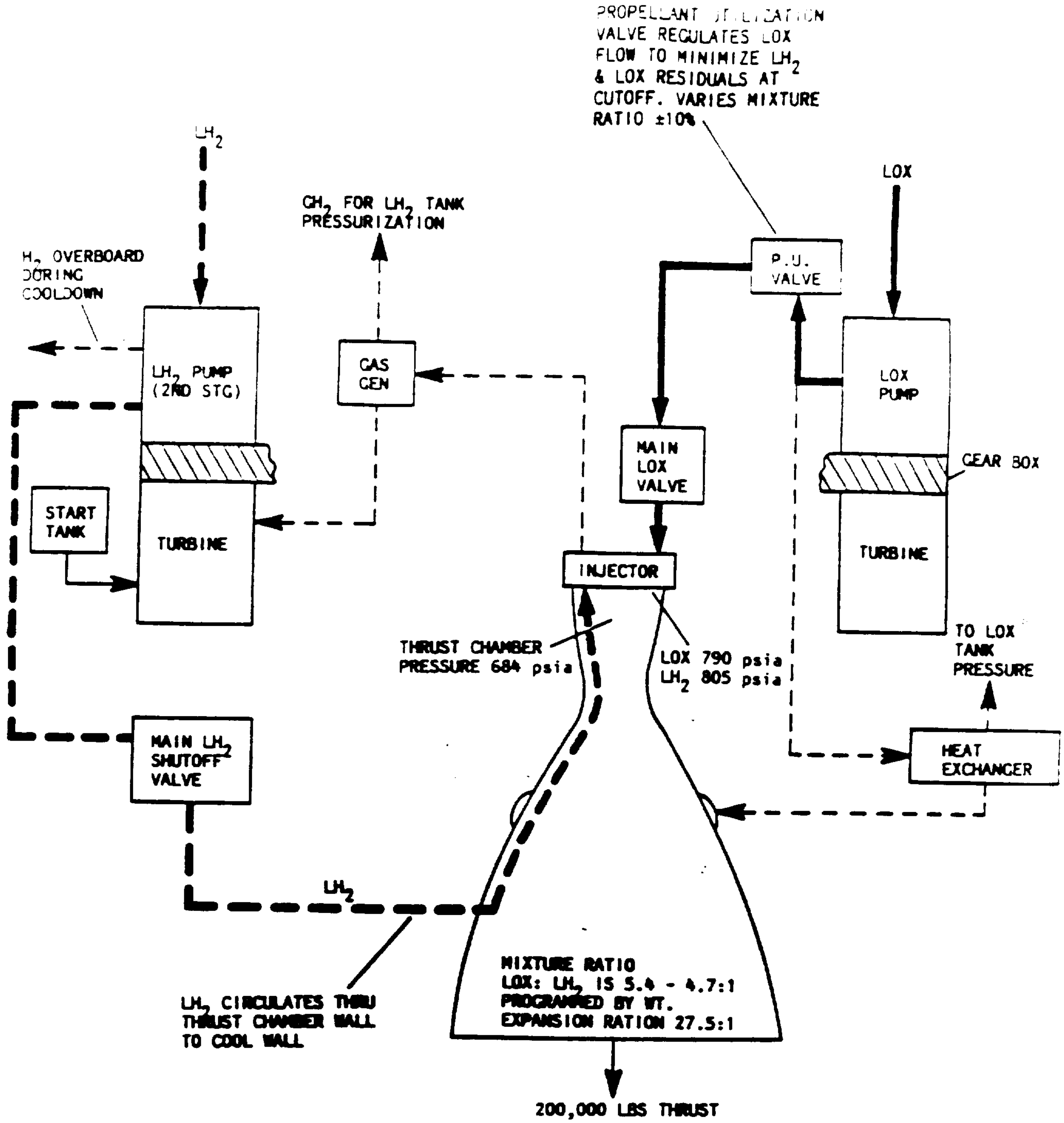


FIGURE 5-6. J-2 ENGINE SYSTEM

- Start tank discharged hydrogen causing the liquid hydrogen and liquid oxygen turbopumps turbines to buildup propellant pressure.

- Liquid oxygen turbopump bypass valve opened to control liquid oxygen pump speed.

- Main oxidizer valve opened allowing liquid oxygen to be injected into thrust chamber.

- GG valves admitted propellants. (Spark ignited propellants causing pressure buildup.)

MAINSTAGE

- OK pressure switches sent mainstage OK signal to CM.

- Engine out lights went out.

- Engine reached and maintained 90 percent thrust or more.

- PU valve controlled mass ratio by returning liquid oxygen from pump discharge to pump inlet.

Propellant combustion was initiated within a cavity called the augmented spark igniter (ASI) which was located in the body of the main injector plate. Ignition was provided by spark plugs. Tank head provided propellant supply pressure. An engine mounted start tank precharged with hydrogen gas provided starting power for the turbopumps.

A self contained pneumatic system was used to actuate all engine valves. The system was comprised of a 1000 cubic inch storage bottle, regulator package, and low pressure accumulator. An electrical control assembly provided control of valve sequencing during start/cutoff, spark plug excitation and internal emergency detection. The mixture ratio control valve provided control of combustion mixture ratio as well as thrust control. The valve was mounted on the liquid oxygen pump and was pneumatically actuated by an engine mounted stage controlled solenoid. The valve recirculated liquid oxygen from the pump outlet manifold back through the inlet.

The four outboard engines were gimbal mounted to allow thrust vector control. A hydraulic pump mounted in the liquid oxygen turbopumps shafts provided hydraulic power for the stage engine actuation system.

S-II PROPULSION SYSTEM TEST OBJECTIVES

Verification of the S-II configuration required extensive testing that included separate system and component tests. The assemblies for propulsion system testing included the Battleship stage at the Santa Susana Field Laboratories and

the All Systems Test Stage (S-II-T) at the Mississippi Test Facility. Additional test articles used in the S-II program are listed below.

The S-II-S configuration was a structurally complete stage used to certify the integrity of the entire airframe under simulated critical loads and pressure differentials.

The S-II-F/D configuration was used at KSC to verify its compatibility with other launch vehicle stages, ground support equipment, the launch pad, and other supporting facilities of launch complex 39. At MSFC the vehicle was used in the Saturn V dynamic testing to determine the dynamic characteristics of the launch vehicle.

A listing of Battleship and S-II-T test objectives follows. Fifteen Battleship and seven S-II-T firings before the first flight were required to satisfy these objectives, although total firing attempts for the total program were sixty-three with twenty-eight successful completions.

- Verify the capability of the stage to accept and deliver cryogenic propellants
- Evaluate performance of the integrated GSE and stage systems
- Verify thermal and structural adequacy of the stage insulation
- Verify acceptable operation of the liquid oxygen anti-geyser system
- Verify operation of the liquid hydrogen and oxygen recirculation systems
- Verify operation of the propellant utilization system for automatic replenishing and control of the engine mixture ratio
- Evaluate performance of the thrust vector control system
- Evaluate flight instrumentation and telemetry system
- Evaluate and calibrate the liquid oxygen depletion cutoff system
- Evaluate the performance of J-2 engines in the clustered configuration
- Determine structural capability of stage systems
- Evaluate the performance of the stage flight propellant pressurization systems

S-II TEST ACCOMPLISHMENTS

A summary of significant accomplishments experienced during S-II stage propulsion system testing is presented in the following paragraphs. Again each

issue has been classified as to consequence and time phasing for which the issue may apply as was done previously in Appendix 3 for Space Shuttle.

Actuator Abnormal Motion (unworkable, preflight/flight). For four to five seconds after "slam" release during S-II-3 static firing, the Engine 3 yaw actuator exhibited abnormal piston position output. The cause was attributed to noise induced on the actuator command by an improper instrumentation circuit. The command current measuring resistor was incorrectly installed. The resistor was relocated and the problem did not recur.

Faulty Solder Joint/Seals (unworkable, preflight/flight). Numerous pressure switch failures were attributed to defective (porous) solder seals. Although a number of corrective actions were taken to eliminate the problem, it was concluded that the solder joint seal was not acceptable for sealing vacuum cavities. Welded cavity seals were implemented and acceptability confirmed in propulsion system testing.

Fill-and-Drain Valve and Recirculation Valve Position Indicator Anomalies (unworkable, preflight). During checkout of the liquid oxygen fill-and-drain valve on S-II-6 at MTF, the "OPEN" indication was not received and could not be corrected by subsequent trouble-shooting adjustment. The same type of problem had been experienced on the recirculation valves and early prevalves which incorporated a similar magnetically actuated position indication device. The failure of the closed or open indication to pick up when the valve was actuated was attributed to an apparent shift in the indicator dead band between ambient calibration and usage at cryogenic testing, and by tolerance on the initial switch setting. The primary concern was that malfunctions of the position indicators on the fill valves would cause propellant loading reverts, holds, and/or scrubs during static firing or launch countdowns due to interlocks. SCD's were revised, quality control procedures were improved, and electrical interlock requirements were deleted/revised. These actions were proper and adequate, although not necessarily comprehensive in that an effective complete redesign was not feasible and probably not possible because of the time required for an effective solution to the problem.

Vent Valve Position Indicator Anomalies (unworkable, preflight). Numerous vent valve position indication anomalies occurred throughout the S-II program on flight stages, test stages, qualification testing, and acceptance testing. The anomalous conditions consisted of dual indications ("open" and "closed" indications both received), erratic indicators, slow or no response, and chatter. The problem was attributed to freezing of air or nitrogen within the switch causing binding of the actuation mechanism. Fixes were made; however, anomalies continued to occur.

Liquid Oxygen Vent Valve Low Reseat (unworkable, preflight). During prestatic firing tests on the S-II-1 stage, a liquid oxygen tank vent valve failed to reseat within the specified pressure range. This was found to be caused by pressure oscillations in the vent valve portion of the pressure-sensing system. The oscillations were eliminated by adding a double-orificed plenum chamber to the

valve pressure sensing system external to the valve main housing. No further problems were encountered.

Liquid Oxygen Tank Mast Percolation (unworkable, flight). During S-II-4 static firing at MTF, percolation of liquid oxygen tank mast caused a premature shutdown. Investigation revealed that presence of liquid in the annular section of the liquid oxygen tank pressurization mast caused an erroneous ullage pressure indication which ultimately led to termination of the firing. Resolution included hardware and operational changes to prevent the entrance of liquid into the mast annular section and to obtain accurate ullage pressure measurements.

Liquid Oxygen Fill and Drain Line Cracked Bellows (unworkable, flight). Post firing inspection of S-II-2 at MTF detected a cracked pressure carrier bellows in the ·· ·t ball joint of the liquid oxygen fill-and-drain line inboard of the GSE disconnect. Failure was attributed to flow induced vibration of the single ply bellows convolution. Redesign of the line assembly included addition of flow liners to all bellows and three ply bellows replaced all single ply bellows. There were no further problems with flow induced vibration in bellows on the S-II program.

AS-502 (S-II-2) Flight Anomaly (unworkable, flight). During the Apollo 6 (AS-502) mission, the S-II-2 stage experienced premature shutdown of two adjacent outboard engines (Engines 2 and 3). Post flight investigation attributed the cause of Engine 2 failure to breakage of the fuel line to the augmented spark igniter (ASI). Shutdown of Engine 3 was caused by closure of its liquid oxygen prevalve by the Engine 2 cutoff signal. The most probable cause for this anomaly was the cross-connection of the electrical connectors to the solenoids of liquid oxygen prevalve 2 and liquid oxygen prevalve 3.

Changes included redesign of both the fuel and oxidizer lines to the ASI assembly, revision of the engine checkout program at KSC to include individual prevalve checks, inspection of electrical wire harnesses on subsequent stages to uncover any incorrect or unclear electrical reference designators (ERD), reassignment of ERD as necessary to prevent misconnections, and installation of additional instrumentation to check the redesigned ASI lines.

The J-2 fuel and oxidizer ASI line modifications were extensively tested in cluster firings on the Battleship. Over 1000 seconds of Battleship testing was conducted with the ASI line modifications installed on three engines.

The new ASI lines were subsequently installed on S-II-3 through S-II-15. The engine performance of these stages verified the adequacy of the ASI line design change.

Liquid Oxygen Sump Baffles (improvement, flight). During sump screen replacements on the S-II-5, it was discovered that one Liquid oxygen anti-vortex sump baffle had a tab sheared off. Similarly, two baffles in S-II-3 and one baffle on the Battleship were found to have cracked tabs. The condition was attributed to high-cycle, low-stress fatigue.

Battleship tests successfully demonstrated that the sump baffles could be removed without impairing liquid oxygen drawdown. The go-ahead was given for removal of the sump baffles from flight stages S-II-3 and subsequent.

Liquid Oxygen Recirculation System (unworkable, preflight). The Battleship boattail environment tests conducted at Santa Susana established the liquid oxygen recirculation system design and procedure to be used on all S-II flight stages. Accomplishments included design and installation of an in-flight liquid oxygen helium injection system and stage and engine insulation.

Liquid Oxygen Tank Vent Valve (unworkable, preflight). Liquid oxygen tank vent valve instability during venting and reseating operations were investigated on the Battleship stage. Modifications were made that proved to be stable. The modified liquid oxygen tank vent valves were acceptable for the flight S-II stage.

Depletion Sensors and ECO Time Delay Performance (workable, mod. expected, flight). The maximum safe liquid oxygen and liquid hydrogen depletion cutoff time delays for use on flight vehicles were obtained.

Vibration Study (unworkable, flight). Comparisons were made between single-engine and cluster-engine vibration environments during firings on the Battleship stage to investigate repeated engine tube and component failure during Battleship testing.

Control Line Modifications (unworkable, flight). A series of Battleship firings was conducted to evaluate structural integrity of flight configuration modifications made to the main fuel valve, OTBV control lines, and the ASI liquid oxygen lines on all five Battleship engines. No hardware problems were noted with these changes.

Liquid Oxygen/Liquid Hydrogen Vent Valves (unworkable, preflight). During a 12-hour hold test on the Battleship, liquid oxygen and liquid hydrogen vent valve microswitches malfunctioned repeatedly under cryogenic conditions. Although the valves opened/closed as required, the open/closed indicators in these valves failed.

Twelve-Hour Hold Environment Freeze Test (workable, preflight). A 12-hour hold test was conducted, simulating the KSC environment and using flight-type components. Analysis indicated that the propulsion system and components were capable of withstanding a 12-hour hold.

Liquid Oxygen Prevalve Relief Valve (unworkable, preflight/flight). During liquid oxygen tanking in support of liquid oxygen recirculation test ST-3, a piece of the liquid oxygen prevalve relief valve housing sheared off on Engine 5 liquid oxygen prevalve. Investigation revealed this was caused by contraction of the liquid oxygen tank during chilldown, pulling the liquid oxygen feed duct assembly into the cruciform. The relief valve was deleted from Engine 5 liquid oxygen prevalve.

Liquid Hydrogen Tank Rupture (catastrophic, preflight/flight). The S-II-T test stage was destroyed on May 28, 1966, when the liquid hydrogen tank ruptured during pressurization with helium. Failure resulted from propagation of a small fracture on a raised boss in cylinder 2 at a pressure below design ultimate.

Liquid Hydrogen/Liquid Oxygen Vent Valve Failures (unworkable, preflight/flight). During the S-II-T test program, vent valve failures were numerous, although temporary in nature. The most common malfunction was the need for a long warmup period to open the cold-soaked vent valve after holding an override closed position. New design valves were provided for the flight stages.

RF and Measurement Systems (unworkable, preflight/flight). Noise and random signals were noted in flowmeter and tachometer measurements. A new configuration plug-in module for flow and tachometer measurements were tested on the Battleship stage and produced acceptable flow/tachometer data.

ASI Liquid Oxygen Supply Line Fatigue (catastrophic, preflight/flight). A full duration firing on S-II-T was terminated 196 seconds after reaching mainstage. Engine 5 ASI liquid oxygen supply line ruptured causing a fire in the Engine 5 area, which burned the mainstage OK pressure switch cable, causing the automatic engine cutoff.

APPENDIX 6

SATURN V/SATURN IB S-IVB MAIN PROPULSION SYSTEM

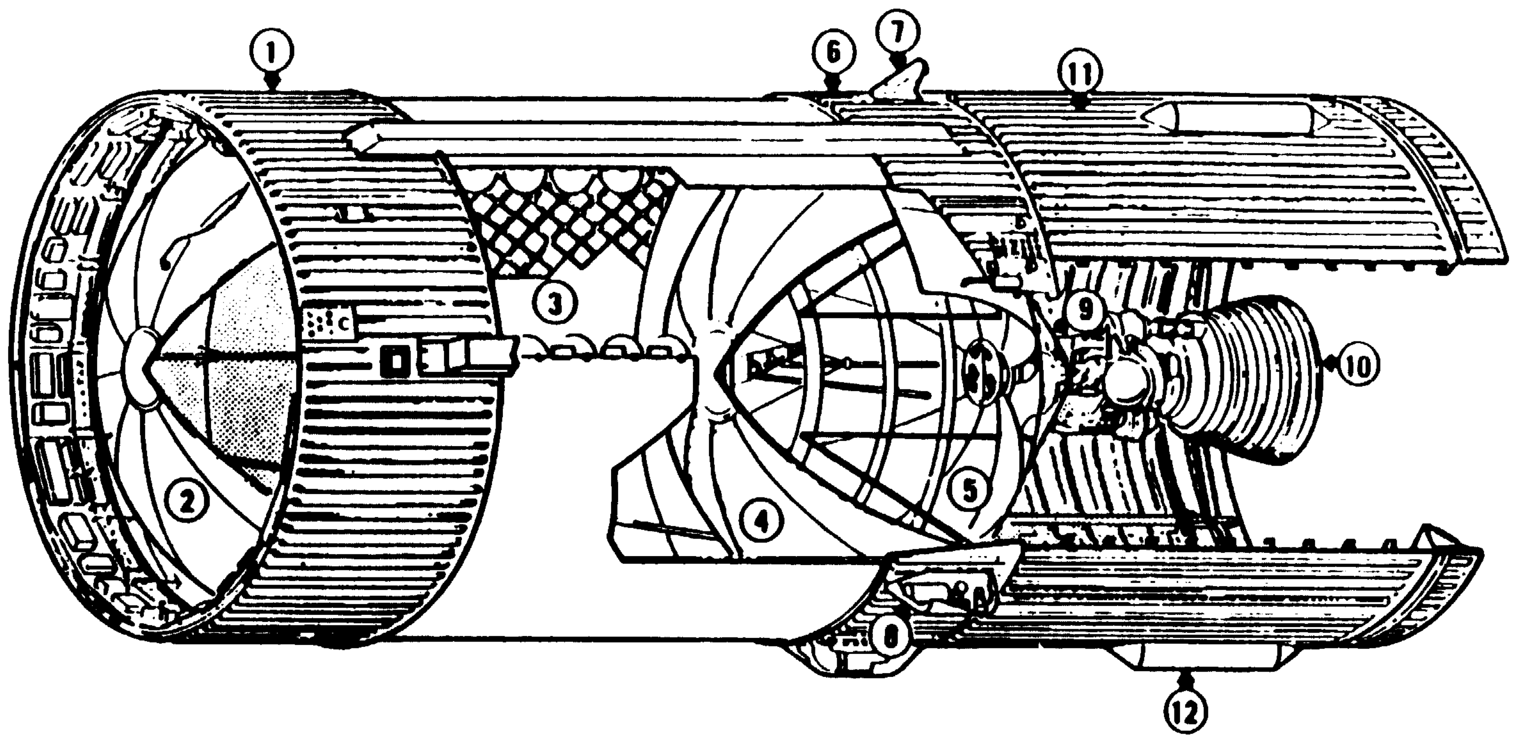
The S-IVB stage was the third stage of Saturn V and the second stage of Saturn IB (Figures 6-1 and 6-2). While the basic stage design for the two missions was the same, several design and operational features differed. The Saturn S-IVB stage with its single J-2 engine represented a unique "slice" of history for liquid cryogenic propulsion systems. Advanced technology was required for the development of such significant areas as engine start/re-start; propellant conditioning, containerization, pressurization, propellant slosh and utilization; thermal conditioning; engine shutdown; vacuum propellant dump; control systems and aborts. The "unknowns and leading-edge" technologies required were many for new stages using new J-2 engines (Figure 6-3) of 200,000 pounds thrust (vacuum)—which were later uprated to 230,000 pounds thrust.

The S-IVB stage consisted of an aft interstage, an aft skirt, a thrust structure, a single gimballed J-2 engine, a divided propellant container, and a forward skirt. The S-IVB/V and S-IVB/IB were similar. However, S-IVB/V translunar excursion mission configuration demanded a heavier payload and required engine restart capability, which necessitated certain structural and system changes. One major structural change was the much larger diameter aft skirt .

Thrust Structure. The thrust structure was an inverted truncated cone attached at its large end to the aft dome of liquid oxygen tank and attached at its small end to the engine mount. The aft dome of the liquid oxygen tank was the lower portion of the propellant container (liquid oxygen and hydrogen tank assembly). The thrust structure provided the attach point for the center line mounted J-2 engine and distributed the engine thrust over the entire tank circumference. Attached external to the thrust structure were the engine piping, wiring, interface panels, eight high pressure helium spheres, hydraulic system, and certain engine and liquid oxygen tank instrumentation.

Propellant Container. The divided propellant container (Figure 6-4) was an internally insulated cylinder with hemispherical bulkheads at each end and a common bulkhead to separate the lower tank of liquid oxygen from the liquid hydrogen tank above. This 21.7 foot diameter common bulkhead was a sandwich type construction consisting of two parallel, hemispherical shaped, aluminum alloy (2104-T6 A1) domes bonded to a fiberglass phenolic honeycomb core (1 3/4"). The internal surface of the liquid hydrogen tank was a milled waffle pattern for tank stiffness and minimum structural weight. To minimize liquid hydrogen boiloff, polyurethane insulation blocks, covered with a fiberglass sheet and coated with a sealant, were bonded into the milled areas of the waffle patterns in the liquid hydrogen tank. The liquid hydrogen tank volume was 10,418 cubic feet and the liquid oxygen tank volume was 2,830 cubic feet.

Forward Skirt. The forward skirt was a longitudinally stiffened cylindrical structure which attached to the forward end of the propellant container and



1	FORWARD SKIRT ASSEMBLY	7	ULLAGE ROCKET MOTOR
2	FUEL TANK FORWARD DOME	8	APS MODULE
3	FUEL TANK INTERIOR	9	THRUST STRUCTURE ASSEMBLY
4	COMMON BULKHEAD	10	J-2 ENGINE
5	OXIDIZER TANK INTERIOR	11	AFT INTERSTAGE ASSEMBLY
6	AFT SKIRT ASSEMBLY	12	S-IB RETRO ROCKET

FIGURE 6-1. SATURN IB S-IVB STAGE

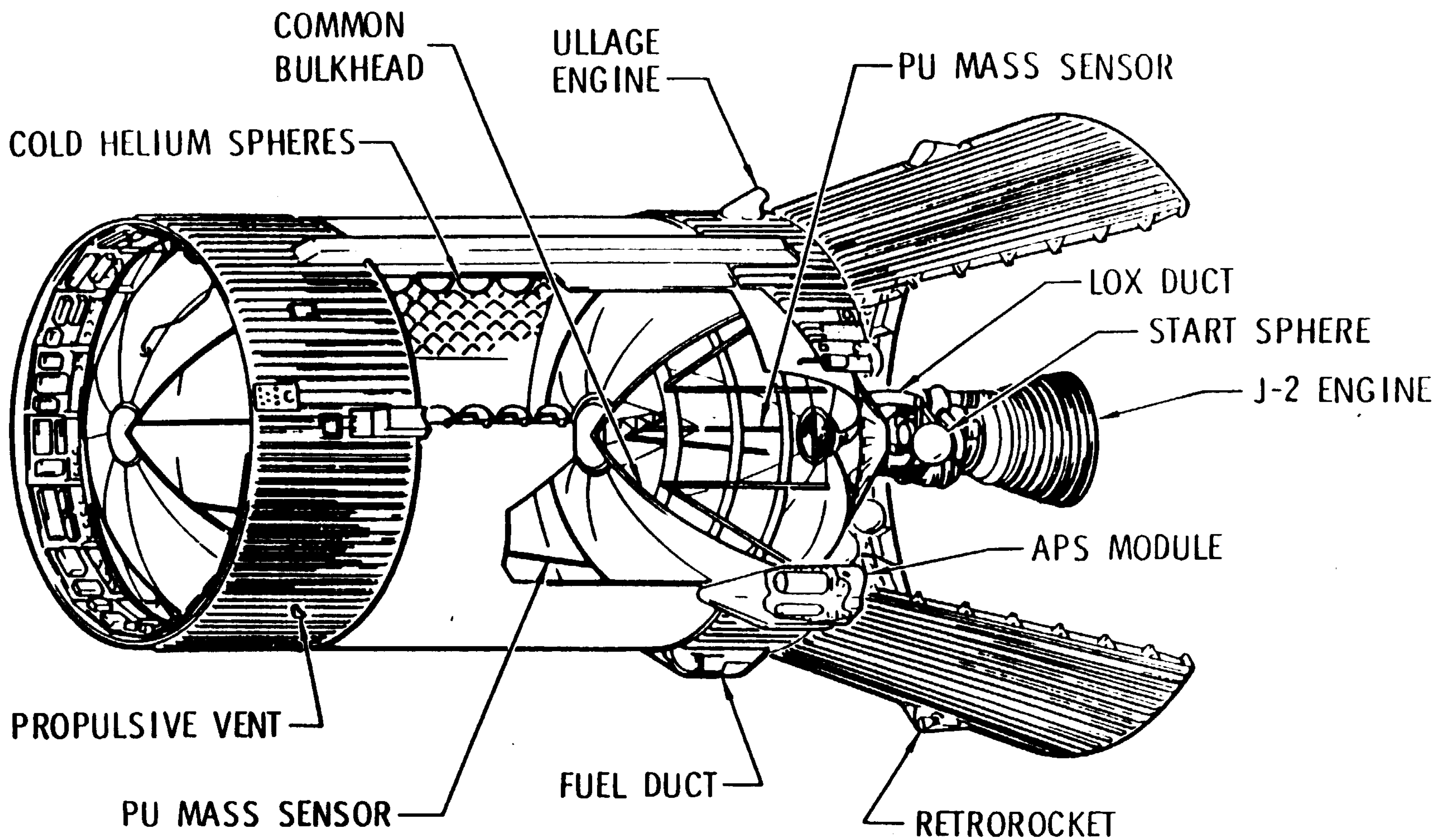


FIGURE 6-2. SATURN V S-IVB STAGE

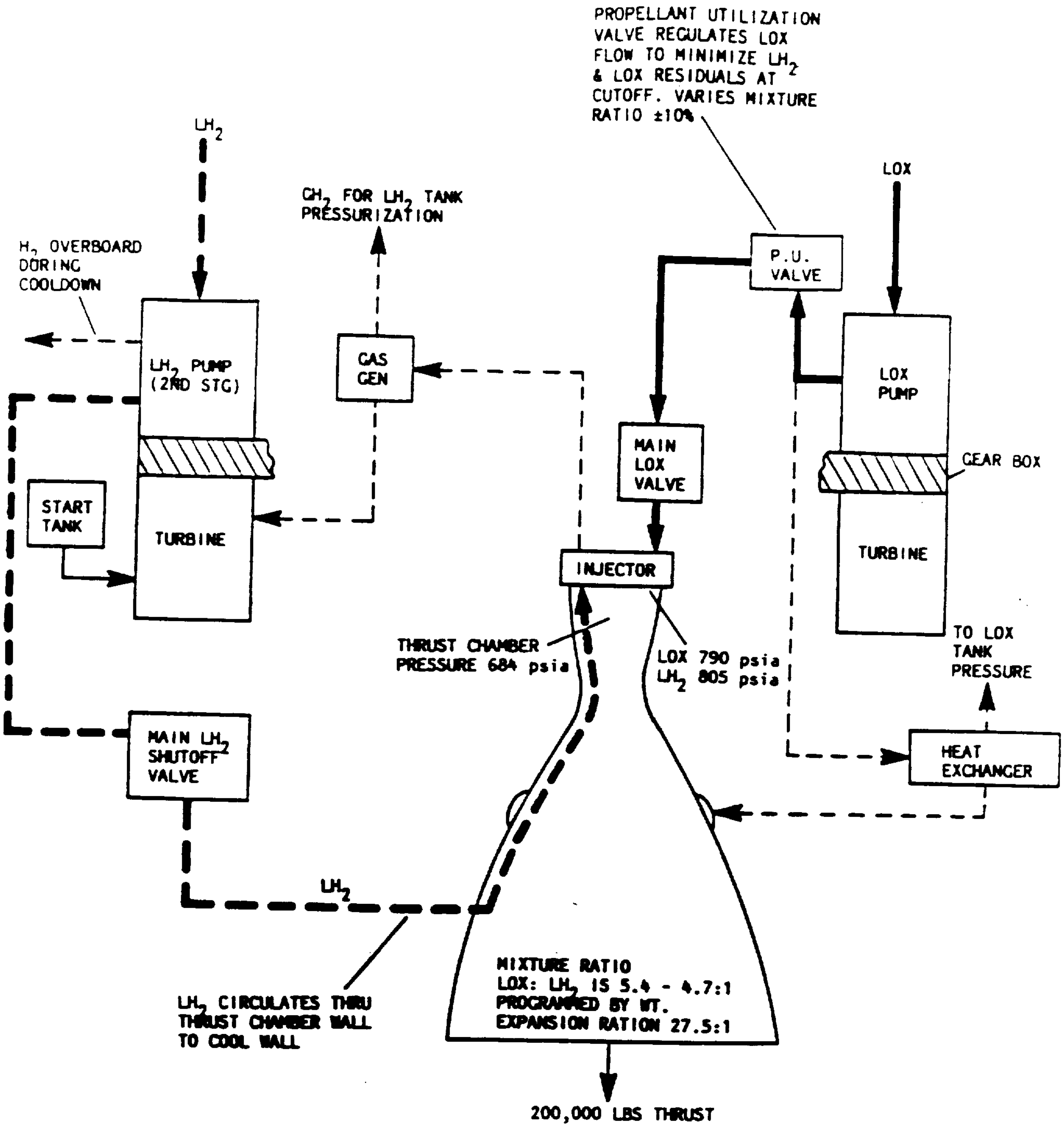


FIGURE 6-3. J-2 ENGINE SYSTEM

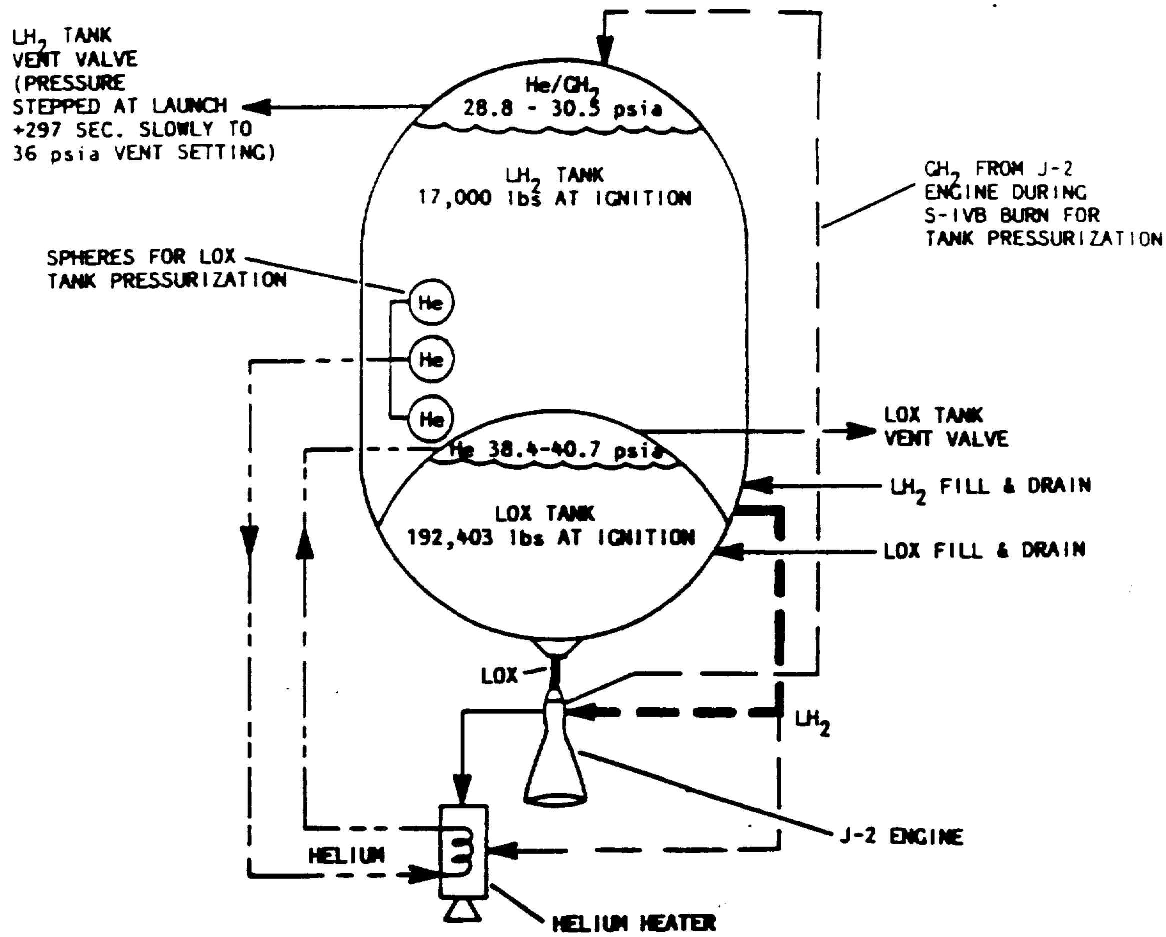


FIGURE 6-4. S-IVB STAGE PROPELLANT SYSTEM

supported the vehicle instrument unit and the Saturn Apollo payload. For safety, the forward skirt area was purged with nitrogen from the vehicle instrument unit while propellants were being loaded or stored in the stage.

Aft Interstage. The aft interstage for Saturn IB was a semimonocoque structure which supported the 21.7 foot diameter S-IVB second stage, the Saturn IB instrument unit, and the Apollo Spacecraft payload prior to first stage separation. The cylindrical aft interstage provided mounting facilities for four of the Thiokol 30,000 pound thrust solid propellant retromotors with aerodynamic fairngs. Upon detonation, these retro motors provided a negative acceleration to the Saturn IB at the aft interstage/aft skirt interface.

S-IVB/IB Aft Skirt. The S-IVB/IB aft skirt was a semimonocoque structure which attached to the aft end of cylindrical portion of the propellant container and provided mounting hardware for three ullage motors. Three Thiokol 3390 pound nominal thrust solid propellant motors (TX-280) were equally spaced to provide a positive acceleration of the second stage to settle the liquid oxygen and hydrogen propellants for the J-2 engine one time start for Saturn IB usage. Two diametrically opposed attitude control modules were attached to the aft skirt. Each of these attitude control modules contained the three TAPCO 150-pound thrust (vacuum), hypergolic (MMH and N_2O_4) rocket engines.

S-IVB/V Aft Skirt. The S-IVB/V aft skirt was a truncated cone that provided the load support structure between the S-IVB stage and the 33-foot diameter S-II stage. The structure was of aluminum skin panels externally stringer-stiffened, riveted construction bolted to the S-IVB aft skirt and the S-II forward skirt. The aft skirt provided the focal point for the required electrical and mechanical interface and an envelope for the aft environmental control. Retrorocket motors were attached to this interstage and at separation, the interstage remained attached to the S-II stage.

J-2 Engine. The S-IVB/IB single burn of the single J-2 engine, with 200,000 pound thrust, occurred immediately after the S-IB stage burn was completed and S-IVB/S-IB stage separation. The S-IVB/IB placed the vehicle instrument unit (IU) attached to the Apollo payload into orbit. Similarly, the S-IVB/V first burn of the J-2 engine occurred immediately after S-II/S-IVB stage separation and lasted long enough to insert the vehicle into earth orbit. Approximately two to four hours later, the second burn of the J-2 engine injected the S-IVB instrument unit/spacecraft into a translunar trajectory. During burn periods, the vehicle control was provided by the J-2 engine which gimballed ± 7.0 degrees in a square pattern powered by a hydraulic actuator control system.

The J-2 engine was a high performance multiple restart engine using liquid oxygen and liquid hydrogen as propellants. The engine attained a thrust of approximately 200,000 pounds. The only substances used in the engine were the propellants and helium gas. The extreme low operating temperatures prohibited the use of lubricants or other fluids. The engine featured a single, tubular-walled, bell-shaped thrust chamber and two independently driven, direct drive, turbopumps for liquid oxygen and liquid hydrogen. Both turbopumps were powered in series by a single gas generator, which used the same propellants as the thrust chamber.

The main hydraulic pump was driven by the oxidizer turbopump turbine. The ratio of fuel to oxidizer was controlled by passing liquid oxygen from the oxidizer turbopump to the inlet side through a servo valve.

The engine valves were controlled by a pneumatic system powered by gaseous helium which was stored in a sphere inside the start bottle. An electric control system, which used solid state logic elements, was used to sequence the start and shutdown operations.

During the burn, the liquid oxygen tank was pressurized by flowing cold helium through the heat exchanger in the oxidizer turbine exhaust duct. The liquid hydrogen tank was pressurized during burn periods by hydrogen from the thrust chamber fuel manifold.

The restart of the S-IVB/V J-2 engine was identical to the initial start.

Helium Storage. Helium was stored in nine cold helium spheres located in the liquid hydrogen tank, eight ambient helium spheres mounted on the thrust structure, and one sphere located inside the start bottle on the engine. The nine cold helium spheres supplied cold helium for pressurization and repressurization of the liquid oxygen tank and repressurization of the liquid hydrogen tank. Five of the eight ambient helium spheres provided an alternate source of helium for repressurization of the liquid hydrogen tank, two provided an alternate source of helium for repressurization of the liquid oxygen tank, and one provided pressure for operation of the stage pneumatic controls. The engine could control which helium sphere provided pressure for operation of the engine controls.

Liquid Oxygen System. Liquid oxygen was stored in the aft tank of the propellant tank structure (Figure 6-5) at a temperature of -297 degrees F. Total volume of the tank was approximately 2830 cubic feet with an ullage volume of approximately 108 cubic feet. The tank was prepressurized between 38 and 41 psia and was maintained at that pressure during boost and engine operation. Gaseous helium was used as the pressurizing agent.

The liquid oxygen fill-and-drain valve was capable of allowing flow in either direction for fill or drain operations. During tank fill, the valve was capable of flowing 1000 gpm of liquid oxygen at an inlet pressure of 51 psia. Pneumatic pressure for operating the fill-and-drain valve was supplied by the stage pneumatic control bottle.

Pressure switches were used to control the tank pressure during fill. In the event of tank overpressurization (41 psia) the pressure switch sent a signal to close the liquid oxygen ground fill valve.

Liquid oxygen tank pressurization (Figure 6-6) was divided into three basic procedures. These procedures were called prepressurization, pressurization, and repressurization. The term prepressurization was used for that portion of the pressurization performed on the ground prior to liftoff. The term pressurization was

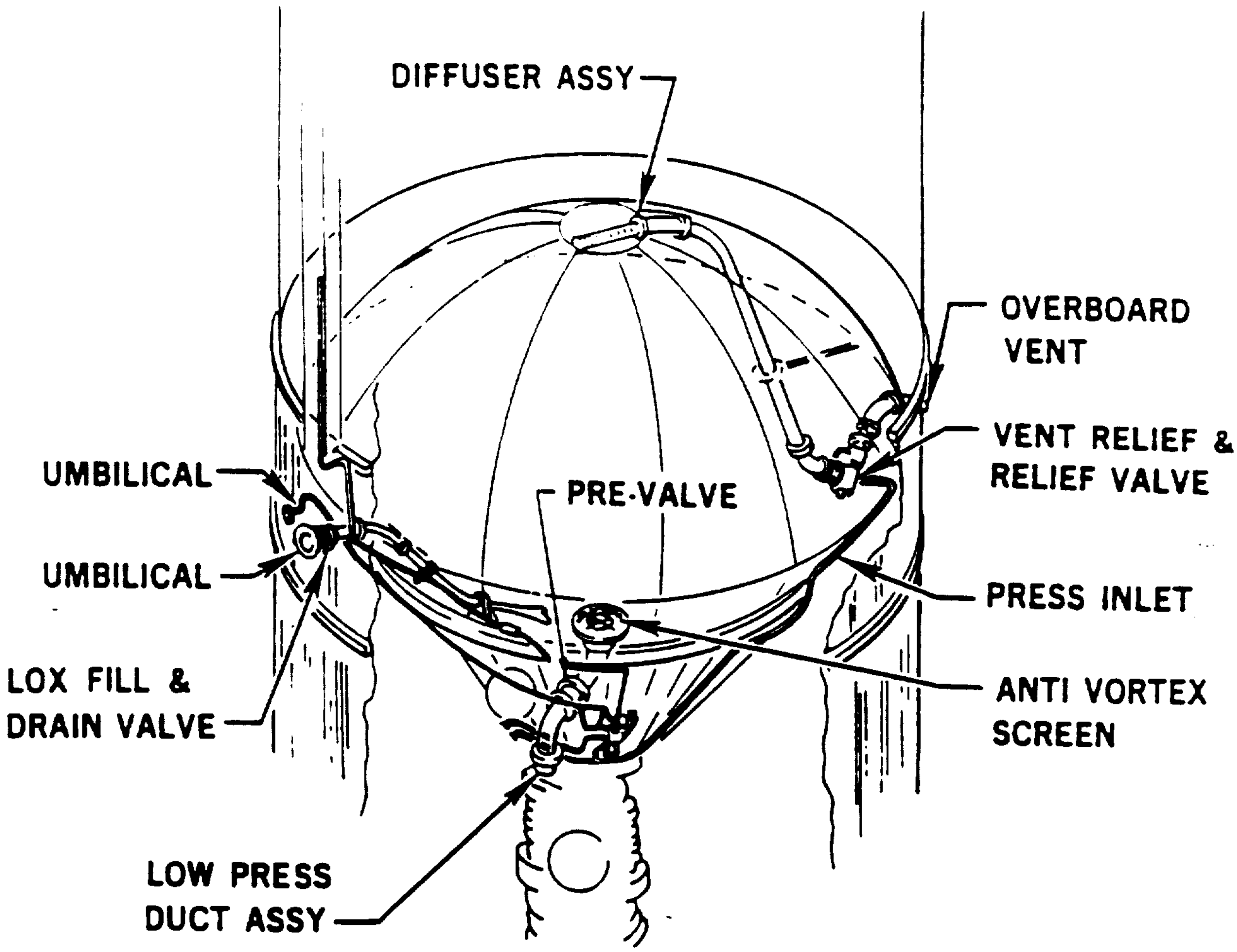


FIGURE 6-5. S-IVB STAGE LOX SYSTEM COMPONENTS

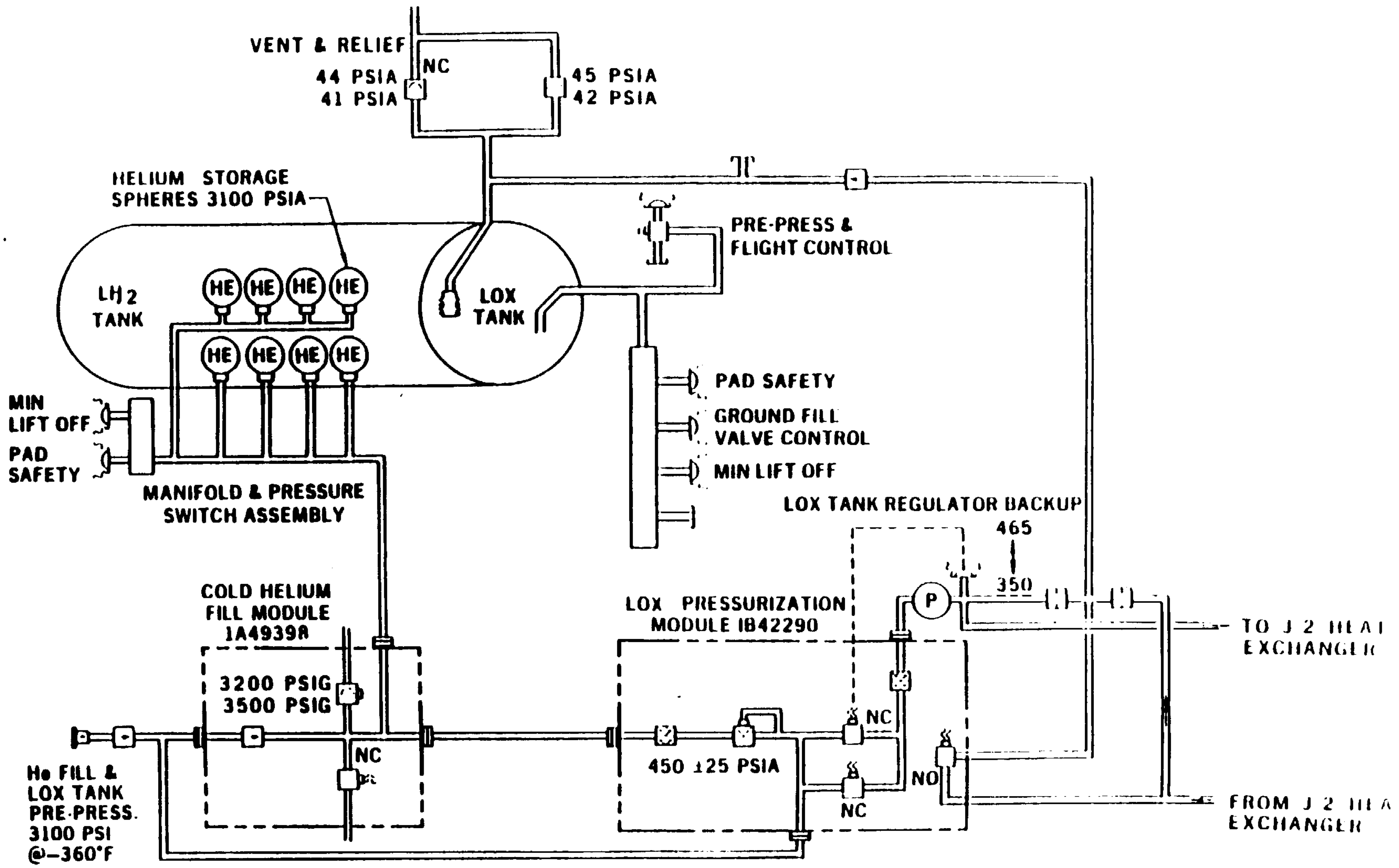


FIGURE 6-6. S-IVB STAGE LOX PRESSURIZATION SYSTEM

used to indicate pressurization during engine burn periods, and lastly, repressurization indicated pressurization just before a burn period.

The pressurant used during the three liquid oxygen tank pressurization procedures was gaseous helium. Cold helium from a ground source was used during the prepressurization period. This ground source of cold helium was also used to charge the nine cold helium storage spheres. The cold helium storage spheres, located in the liquid hydrogen tank, supplied cold helium for both the pressurization and repressurization periods. A hydrogen/oxygen burner was used to heat helium gas supplied from the cold storage spheres. The ambient helium storage spheres, filled by ground support equipment, were the alternate source of helium for use during repressurization prior to the first and second burns.

The liquid oxygen tank pressure was controlled by the flight control pressure switch regardless of the pressurization procedure used. These switches controlled solenoid shutoff valves in each of the supply subsystems.

The liquid oxygen tank vent subsystem provided for controlled liquid oxygen tank venting during normal stage operation and for pressure relief venting when tank overpressures occurred. The liquid oxygen tank venting subsystem included a propulsive venting vent and relief valve and a latch open non-propulsive vent valve. The vent and relief valve was pneumatically operated upon receipt of a ground command.

Successful engine start required the liquid oxygen pump and related hardware to be at liquid oxygen temperature and that all oxygen available for consumption be single phase and satisfy specific requirements. This was accomplished by a small pump and valve in a fluid circuit about the feedline prevalve. With the prevalve closed, oxygen from the tank was forced through the feedline, the engine, and a small return line connecting the engine and oxygen tank. The system operated prior to launch and during powered flight until engine ignition. For the orbital restart mission, a similar sequence was used, however, pumping was not initiated prior to ullage rocket firing. This ensured continued oxygen availability to the small pump.

Liquid Hydrogen System. The liquid hydrogen (Figure 6-7) was stored in an insulated tank with a total volume of approximately 10,400 cubic feet with an ullage volume of approximately 300 cubic feet. The liquid hydrogen tank was prepressurized to 28 psia minimum and 31 psia maximum.

Prior to loading, the liquid hydrogen tank was purged with helium gas. At the initiation of loading, the ground controlled combination vent and relief valve was opened, and the directional control valve was positioned to route hydrogen overboard to the burn pond.

Liquid level during fill was monitored by means of liquid hydrogen mass probes. A backup overfill sensor was provided to terminate flow in the event of a 100 percent load cutoff failure. Pressure switches controlled the tank pressure

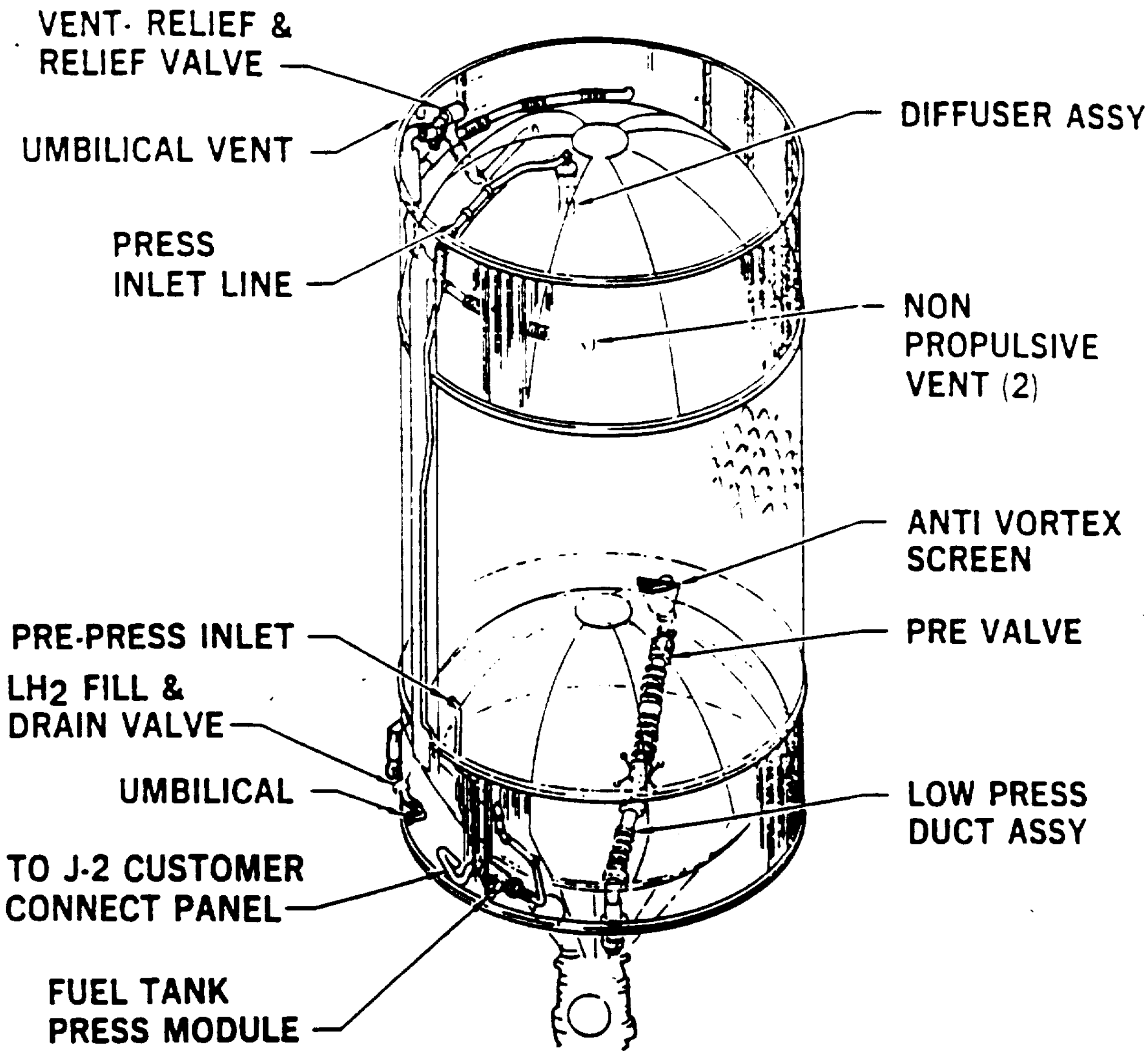


FIGURE 6-7. S-IVB STAGE FUEL SYSTEM COMPONENTS

during fill. In the event of tank over pressurization (31 psia), the pressure switch sent a signal to close the liquid hydrogen ground fill valve.

Liquid hydrogen tank pressurization (Figure 6-8) was also divided into prepressurization, pressurization, and repressurization procedures similar to the liquid oxygen tank.

The pressurants used during the three liquid hydrogen tank pressurization procedures were gaseous hydrogen (hydrogen) and gaseous helium. Cold helium from a ground source was used during the prepressurization period. The cold helium storage spheres (2), located in the liquid hydrogen tank supplied cold helium for use during the repressurization period. The five ambient helium storage spheres filled by ground support equipment supplied an alternate source of helium for use during the repressurization period.

The liquid hydrogen tank pressure was controlled by the flight control pressure switch regardless of the pressurization procedures used. This switch controlled solenoid shutoff valves in each of the supply subsystems.

The liquid hydrogen tank vent subsystem was equipped to provide either propulsive or non-propulsive venting. Non-propulsive venting was the normal mode used, except for the Saturn V S-IVB orbital mission. During the two to four hours the vehicle was in earth orbit, excluding approximately five minutes prior to engine start, hydrogen gas from the tank was vented propulsively. This was necessary to keep hydrogen "settled" to avoid venting liquid hydrogen, thus preserving hydrogen for the engine second burn mission.

Similarly, as for oxygen, successful engine start required a cold engine hydrogen pump and related hardware and acceptable propellant for consumption. A similar system to the oxygen system was used.

S-IVB PROPULSION SYSTEM TEST OBJECTIVES

The major objectives of the S-IVB stage for both Saturn V and IB missions are presented. Both battleship and all system vehicle configurations were involved, and the objectives apply to each as appropriate. There were a total of 24 firing attempts, excluding altitude testing, which was conducted in a test chamber located at the Arnold Engineering Development Center and later S-IVB testing at the MSFC. Test objectives follow:

- Determine structural acceptability of engines and vehicle hardware in the propulsion system test environment

- Determine functional capability of engines and vehicle hardware in the propulsion system test environment

- Validate engine-imposed vehicle design requirements

- Validate vehicle/propulsion system imposed engine requirements

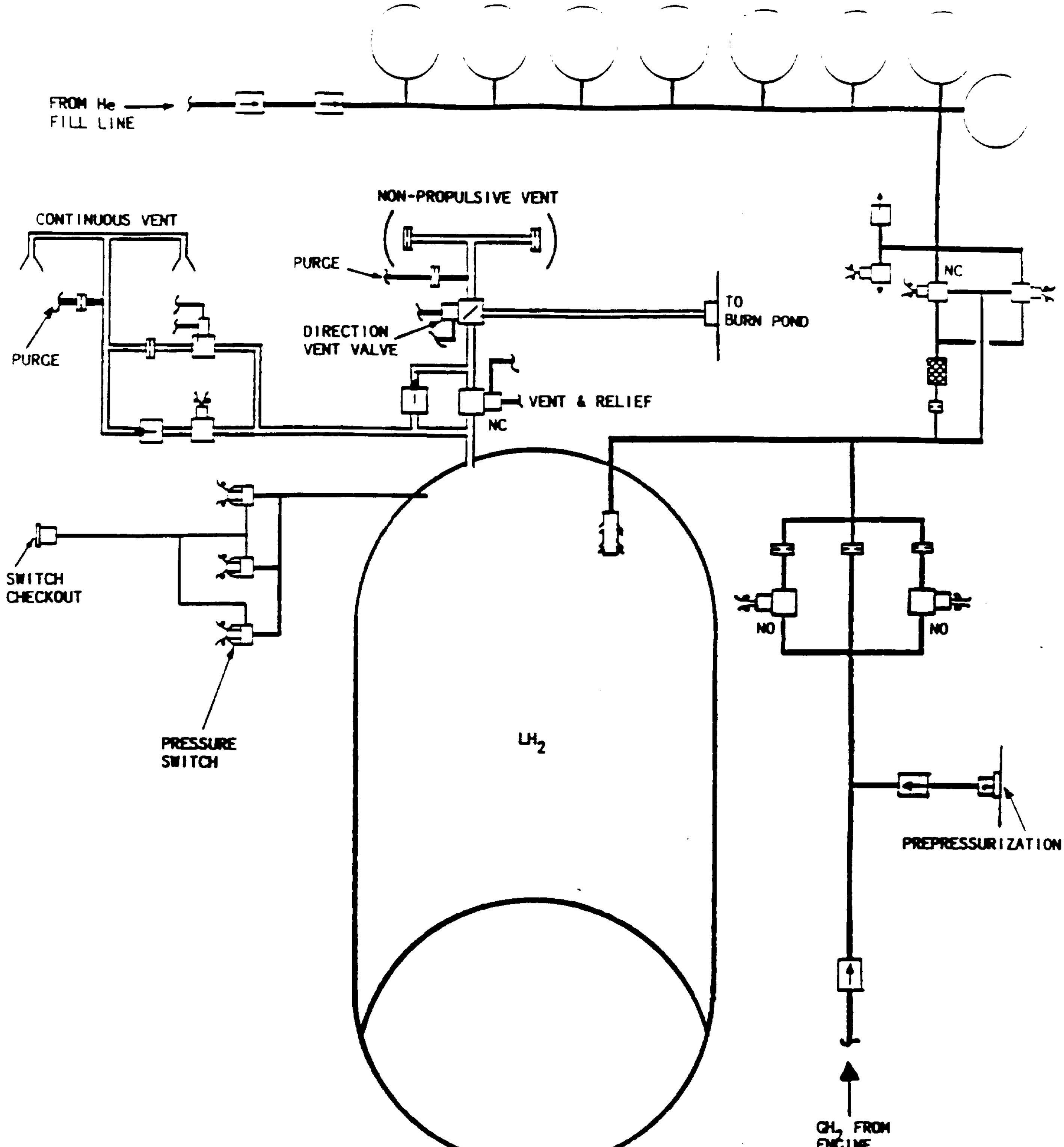


FIGURE 6-8. S-IVB STAGE FUEL PRESSURIZATION SYSTEM

- Determine the engine and vehicle hardware propellant leakage integrity
- Validate the functional capability of the engine and vehicle electrical hardware and instrumentation
- Determine functional compatibility of vehicle structure, fluid, electrical, and hydraulic systems with the ground/GSE interfacing systems
- Establish adequacy of operating procedures
- Verify thermal and structural adequacy of the stage insulation
- Demonstrate acceptable operations and adequacy of all systems to satisfy requirements

S-IVB TEST ACCOMPLISHMENTS

Following is a list of accomplishments achieved by the S-IVB propulsion system test program. Each issue has been classified as to consequence and time phasing for which the issue may apply, as was done in Appendix 3 for Space Shuttle.

<u>Recirculation System</u>

<u>Recirculation Pump Destruction (unworkable - preflight)</u>. Electrically driven vehicle located pumps, one oxygen and one hydrogen pump, were utilized to circulate small quantities of each propellant through the engine and respective feedline to remove heat and provide acceptable quality propellants in each feedline at engine start. During tank inerting prior to propellant loading and tank blowdown post firing, these pumps were frequently run as a turbine (wrong direction) at excessive speeds thus damaging pump bearings and inverters and necessitating hardware changeout. This required addition of protective braking devices to the pumps and extensive changes to procedures.

<u>Liquid Hydrogen Pump Chill Procedure (catastrophic - flight)</u>. Early battleship testing indicated the potential for liquid hydrogen pump stall during the initial phase of J-2 engine start. Engine start for S-IVB/1B and first start for the S-IVB/V was approximately 2 and 11 minutes after liftoff respectively. S-IVB/V was restarted several hours after liftoff . Potential consequences were that the engine would fail to start and mission failure would result. Turbopump chill procedures were changed and validated.

<u>Thrust Chamber Chill Down (catastrophic - flight)</u>. Early battleship firings satisfied the engine supplier's thrust chamber start requirement of 260 $\pm$ 50 degrees Rankin; however, further testing discovered that the requirement was inadequate to assure a satisfactory start in all circumstances. Extensive additional tests were necessary to explore a number of other variables to assure satisfactory start in the various vehicle environments. Some variables were liquid hydrogen lead time, helium supply quantities and supply conditions.

System Evaluation of S-IVB Liquid Hydrogen/Liquid Oxygen Engine and Feedlines Chilldown Systems (catastrophic - flight). The performance of the systems provided for chilldown of the engine and feed ducting for both the liquid hydrogen and oxygen propellants were of utmost importance to stage operations and mission success. Performance of each system was sensitive to the environment, to stored energy of the appropriate vehicle lines and the engine, and to the quality of the S-IVB stage and J-2 engine propellant line and pump insulations which controlled heat flow to the engine. Each design utilized electric driven pumps as the fluid "driver". Systems were required to operate at gravity levels of one, greater than one and significantly less than one thus complicating design and verification. Extensive system testing was necessary to support detailed analysis for the S-IVB initial start and particularly the restart mission which occurred near zero gravity. A special orbital flight experiment was necessary to confirm acceptable system operation along with other test objectives. Specific issues to resolve were many: (1) removal of sufficient engine heat; (2) maintain only liquid propellant within the feedline at engine start; and (3) accomplish the above without reorienting propellant within tankage.

Pressurization System

Liquid Hydrogen Pressurant Diffuser (catastrophic - flight). The initial pressurant diffuser design for the hydrogen tank of S-IVB/1B and V resulted in tank pressure decay during tests. The diffuser created excessive gas velocities at the liquid gas interface which reduced/destroyed the stratification layer near the liquid surface. The diffuser was enclosed within a woven nylon bag which uniformly diffused the high velocity gas and resolved the issue. To achieve a structurally sound bag, more than one bag and test was required.

Oxygen/Hydrogen Burner Integration (catastrophic - flight). The oxygen/hydrogen burner was utilized to heat cold helium gas for repressurizing the liquid oxygen and hydrogen tanks for approximately 8 minutes prior to S-IVB/V restart in earth orbit. Problems were encountered during component development and significant problems were experienced during integration and system verification. A similar device termed "the helium heater" had been previously developed and used on the S-IV stage for oxygen tank pressurization during powered flight and many of the same/similar problems were experienced. For S-IVB/V, ignition occurred near zero gravity in a vacuum and in a thermal environment which could only be estimated. Quantities of helium gas required were influenced by heat transfer mechanisms within the hydrogen and oxygen tanks and the heat transfer from external surroundings to the vehicle which in turn were influenced by vehicle orientation, timing for vehicle start, surface properties, insulation thermal conductivity, etc. Problems encountered in integration are too numerous to discuss herein; however vehicle "system" tests are necessary to identify many of the problems and to collect important data to resolve problems. The alternative to the oxygen/hydrogen burner is many ambient temperature storage containers filled with 4500 PSI helium gas.

Pressurization System Control Modules (unworkable - flight, preflight). Hardware problems necessitating changeout and design change were frequent occurrences for both the hydrogen and oxygen pressurization control modules. The Battleship program identified/resolved many difficulties in the initial design, although flight vehicle acceptance testing experienced numerous additional problems which had to be resolved, and the Battleship, in some instances, assisted in resolving flight vehicle difficulties. A sampling of difficulties experienced follow:

a. The liquid hydrogen module permitted a sudden 400 percent increase in pressurant flow. An orifice became displaced; thus a retaining system redesign was required.
b. The liquid hydrogen module shutoff valve failed to open when required valve performance was affected by the cryogenic temperature environment.
c. The liquid oxygen module shutoff valve failed to close prior to vehicle start thus causing scrubbing of two tests. Excessive valve gate closing velocity damaged the lip seal necessitating redesign.

Side Wall/Bulkhead Insulation

Liquid Hydrogen/Liquid Oxygen Common Bulkhead (unworkable - preflight). S-IVB Battleship utilized an evacuated steel common bulkhead versus the honeycomb structure used for the flight configuration. Exposure to operational pressure and thermal environments of static firings resulted in wrinkles in the face sheet on the liquid hydrogen side of the bulkhead. The flight design initial exposure to environments of all system testing was the S-IB-201 flight vehicle acceptance firing although structural testing had occurred on the structural test article. During acceptance testing a wrinkle in the face sheet on the hydrogen tank side was observed plus local debonding--a strip approximately 1 1/2 inches wide and the length of the wrinkle across one gore, across the bulkhead dollar plate, and into another gore. A similar failure had previously occurred on the S-IV stage common bulkhead. Extensive testing was conducted, including dye penetration, and bulkhead vacuum decay to determine the physical state of the hardware. Finally the S-IVB-201 vehicle was determined usable subject to an acceptable repeat of the bulkhead vacuum decay test at KSC.

Liquid Hydrogen Tank Insulation Debonding (unworkable - preflight). Pressure and thermal cycle exposure during propellant loading and vehicle operation resulted in numerous areas of insulation which were debonded. Such testing foretold of similar concerns for the flight vehicle and the necessity of including schedule time for inspection and repair.

Hydrogen Tank Insulation Thermal Conductivity (unworkable - flight). The stage utilized 3-D reinforced foam tile insulation internal to the hydrogen tank side wall and top bulkhead. A special liner material bonded to the insulation was to separate the foam insulation from the liquid hydrogen. Full scale vehicle tests proved the liner was not impervious to hydrogen and that the insulation thermal conductivity was equivalent to that of hydrogen gas which was significantly greater than pure foam. Extensive testing and analyses were necessary to establish this

fact. While hardware changes were not required, extensive analytical model changes and stage design parameter reassessments were necessary.

Propellant Management

Propellant Management (catastrophic - flight). This subject includes loading a specified propellant quantity, consumption of a specified propellant quantity, maintaining necessary reserves for safe shutdown and mathematically representing rocket engine performance accurately for predicting total vehicle performance. The static firing program was essential for developing and or verifying the involved hardware and procedures. The S-IVB stage utilized active propellant management systems for controlling propellant consumption during flight. These systems utilized mass probes (capacitor) and the probes were also used for loading. Propellant depletion sensors were used for assuring controlled engine shutdown. A quantity of each propellant is not useable for engine consumption for various reasons. This quantity must be minimized and must be known. The quantity of propellant which a tank may store is influenced by accumulative tolerance during construction and dimension of changes resulting from thermal expansion and applied loads. Tank calibration by some means is necessary. Propellant management is thus a broad, complex, and major task which must be developed and carefully verified. The flight vehicle configuration is the best source for development and verification of subject system, analytical tools, and procedures. Problems with development and verification of the propellant management systems were numerous and changes were required in hardware, analysis techniques, and operational procedures. Emphasis on individual issues and problems experienced would be counter productive because of the breadth of the verification activity.

Available Rocket Engine Pump Inlet NPSP (catastrophic, flight). Rocket engine pumps require a minimum NPSP to operate properly. Available NPSP is dependent upon many parameters: (1) tank pressure; (2) propellant temperature; (3) pressure drop of the propellant in the feedlines between the tank outlet and pump inlet; and (4) propellant liquid head. Propellant temperature is dependent upon three variables for a particular propellant: (1) heat entering the propellant; (2) initial temperature of propellant which depends upon propellant tank vent system back pressure and propellant head; and (3) the stratification characteristics of the propellant near the surface at the liquid/gas interface. All parameters are important objectives of any vehicle design and verification program. Stage system testing required procedure changes, hardware design changes, and hardware changeout. Excessive high insulation thermal conductivity required propulsion system re-evaluation. Vent system back pressure is a critical parameter, particularly for hydrogen propellants which must be ducted and burned. This area is always eventful as characteristics of launch site disposal for hydrogen burn ponds differ from burn stacks normally used on verification sites. Feedline pressure loss, while seemingly routine, must be accurately known to verify design estimates and the "end-to-end values" needed are rarely available from other sources. The parameters not discussed in detail are likewise important and must be thoroughly understood.

Miscellaneous

Compartment/Equipment Hazardous Gas and Thermal Control (catastrophic - flight). An engine gas generator fire and explosion occurred at ignition on the Battleship. The cause was gas generator failure due to low hardware temperature resulting from long exposure to surrounding cold hardware and inadequate local compartment purges for thermal control and inerting. This is one issue of many involved in development and system testing for controlling hazardous gas (for engine compartments and inner stage areas) to acceptable concentration levels, systems for monitoring concentration levels, thermal control of propulsion and electronics equipment and necessary thermal measurement systems. Restart from earth orbit after several hours exposure complicates thermal control. A heater blanket was the "fix" for the gas generator problem. Most electronics were mounted on cold plates and temperatures were controlled by active systems, thus avoiding the difficulties of some propulsion hardware.

Liquid Hydrogen Leak Detection (workable - preflight). Propellant leakage in closed or vented compartments is a critical issue. Liquid hydrogen propulsion component leakage on S-IVB Battleship resulted in an explosion and fire during the first S-IVB/V firing. S-IVB/V battleship was a conversion from the S-IVB/1B configuration. Saturn V and S-IVB/V had approximately 7000 and 400 fluid connectors respectively as leak sources. Demonstration of leak free hardware is a prime objective of system testing.

Propulsion System Ducting/Vent Valves (unworkable - flight, preflight). Hardware problems necessitating changeout and design change were frequent occurrences for both the hydrogen and oxygen systems. The Battleship program identified many and assisted in the resolution. Some of the many issues follow:

a. Environmental factors caused liquid oxygen tank vent valve signal irregularities during venting and pressurization functions. Bench tests indicated normal operation. Redesign was required.

b. Environmental conditions caused liquid hydrogen fill-and-drain valve position feedback irregularities during loading and unloading. Bench tests indicated normal operation.

c. The liquid hydrogen feedline inner liner failed, wrinkled, or collapsed, during Battleship testing. Initial failures discovered early in the program were in the upper line section. The duct was redesigned using a triple ply liner. In 1969 during Battleship testing at MSFC, similar failures occurred in lower sections of the feedline. This section had not been redesigned earlier. The lower line failures are life type issues versus basic design deficiencies for upper lines observed early in the program.

d. The liquid hydrogen feed duct utilized vacuum jacketed ducting to reduce heat input to the propellant. The loss of vacuum and thus its thermal insulation characteristics was a frequent occurrence during Battleship testing and early vehicle acceptance firings. Vacuum loss resulted from leaking of

the rupture disk and crack failures of the bellows. The vehicle required frequent servicing to maintain the necessary vacuum, and hardware changeout was often necessary. Continual working with suppliers of the ducting was necessary for satisfactory resolution.

Late Battleship Tests

In late 1968 and 1969, Battleship testing was resumed at MSFC. Primary emphasis was mission enhancement/safety/improvements. Five issues from subject testing are included. For reference, Apollo 8, first manned Saturn V, flew in December 1968 to the vicinity of the moon.

Engine Start Tank Recharge in Orbit (improvement - flight). A leaking engine start tank system could jeopardize successful stage restart from earth orbit. Start tank repressurization from the stage repressurization gas system was investigated and verified.

Engine Structural/Propulsion System Interactions (unworkable - flight). The S-II stage center engine of Flight 503 experienced structural oscillations at a frequency of approximately 18 Hertz. The validity of pump pressure transfer functions which had been used in analyses to determine vehicle stability were of concern, November 1988. A hydraulic driven pulser was installed on the oxygen feedline of the S-IVB Battleship to input pressure oscillation and study transfer functions of the feed system/J-2 engine. Test data revealed the engine manufacturer supplied data, which had been used in structural/propulsion system dynamic analyses, were incorrect. The data were collected to support S-II stage development. Structural/propulsion interaction was not an S-IVB problem.

Helium Pressurization Shut-Off Valve Failure (unworkable - flight). The cold helium pressurization storage system shut-off valve failed to close during the countdown demonstration test in S-IVB-503 at KSC. The replacement pressurizing control module and alternate or backup modules for several vehicles were cryogenically tested on S-IVB Battleship prior to designated usage.

Liquid Oxygen Vent and Relief, Relief Valve Failure Due to Freezing (unworkable - flight). Battleship testing determined thermal operational/hardware problems. Tests were conducted to determine if the liquid oxygen vent and relief valves would both freeze in the closed position or operate in a cyclic mode (freeze-thaw, freeze-thaw, etc.), with failure of the cold helium shut-off valves. Freezing did occur.

Liquid Hydrogen Propellant Tank Drainage (improvement - flight). A liquid hydrogen feed duct collector manifold to improve utilization of residuals was tested in the MSFC S-IVB Battleship. The large screen section of the collector manifold collapsed at specification conditions.

Note: There was no all-systems vehicle for S-IVB. Also, the battleship program for S-IVB/V was extremely short in duration–approximately 1 week. It was shipped to MSFC for some testing, then to Tullahoma, Tennessee for

altitude/environmental testing, then to MSFC for more testing. Because of this series of events, acceptance testing of stages experienced an abundance of failures since they were, of necessity, development vehicles.

APPENDIX 7

SATURN I, S-I STAGE AND SATURN IB, S-IB STAGE MAIN PROPULSION SYSTEM

S-I and S-IB stages were the first stages of the Saturns I and IB launch vehicles respectively (Figure 7-1). Both were designed and developed principally by the MSFC. The S-IB stage was an improved S-I stage primarily to achieve increased performance capability. Because of the similarity of the two designs most development testing was conducted with the S-I stage configuration (Figure 7-2). Figures 7-3 through 7-7 provide insight relative to the stage configuration and important system designs.

The S-I stage was fueled by liquid oxygen/RP-1 fuel. Thrust was provided by eight H-1 engines which initially produced a thrust of 165,000 pounds each. The engines were arranged in a cluster of four inboard and four outboard engines. The four outboard engines could be gimballed in both pitch and yaw to provide thrust vector control.

The stage used clustered cylindrical tanks for fuel and oxidizer storage. The tank arrangement was such that four 70-inch diameter fuel tanks, alternating with four 70-inch diameter oxidizer tanks, were clustered around a 105-inch diameter center oxidizer tank. Each of the four fuel tanks fed two engines, yet were interconnected with the other tanks. The center oxidizer tank provided series flow to the four outboard oxidizer tanks, which also fed two engines apiece. Pressurization of the oxidizer tanks was done by a heat exchanger from each engine which produced gaseous oxygen which was collected and ducted to the top of the center oxidizer tank and subsequently to individual tanks. Gaseous nitrogen from 48 fiberglass spheres mounted at the top of the stage pressurized the fuel tanks. This was later changed to a helium system which included two helium storage bottles mounted to the top of the stage.

The cluster of tanks was held together at the base by the tail section and at the top by a structural component known as the "spider beam". The tail section consisted of the thrust structure assembly as well as the heat shield, shrouding for engine components, holddown points, stabilizing fins (on later S-I and S-IB stages), and other components.

A test booster called SA-T was manufactured for use in the propulsion system test program. The SA-T booster was modified throughout the program to keep it updated to the current flight configuration. As soon as a flight booster completed its acceptance test series and was shipped to Cape Canaveral, the updated SA-T was installed into the test stand for verification of the next flight configuration.

In most respects, the S-IB stage retained the size and shape of its S-I predecessor. The upper area was modified to take the larger diameter and heavier S-IVB stage, and the aerodynamic fins were redesigned for the longer and heavier vehicle. The thrust of the H-1 engines was increased to 200,000 pounds each, and

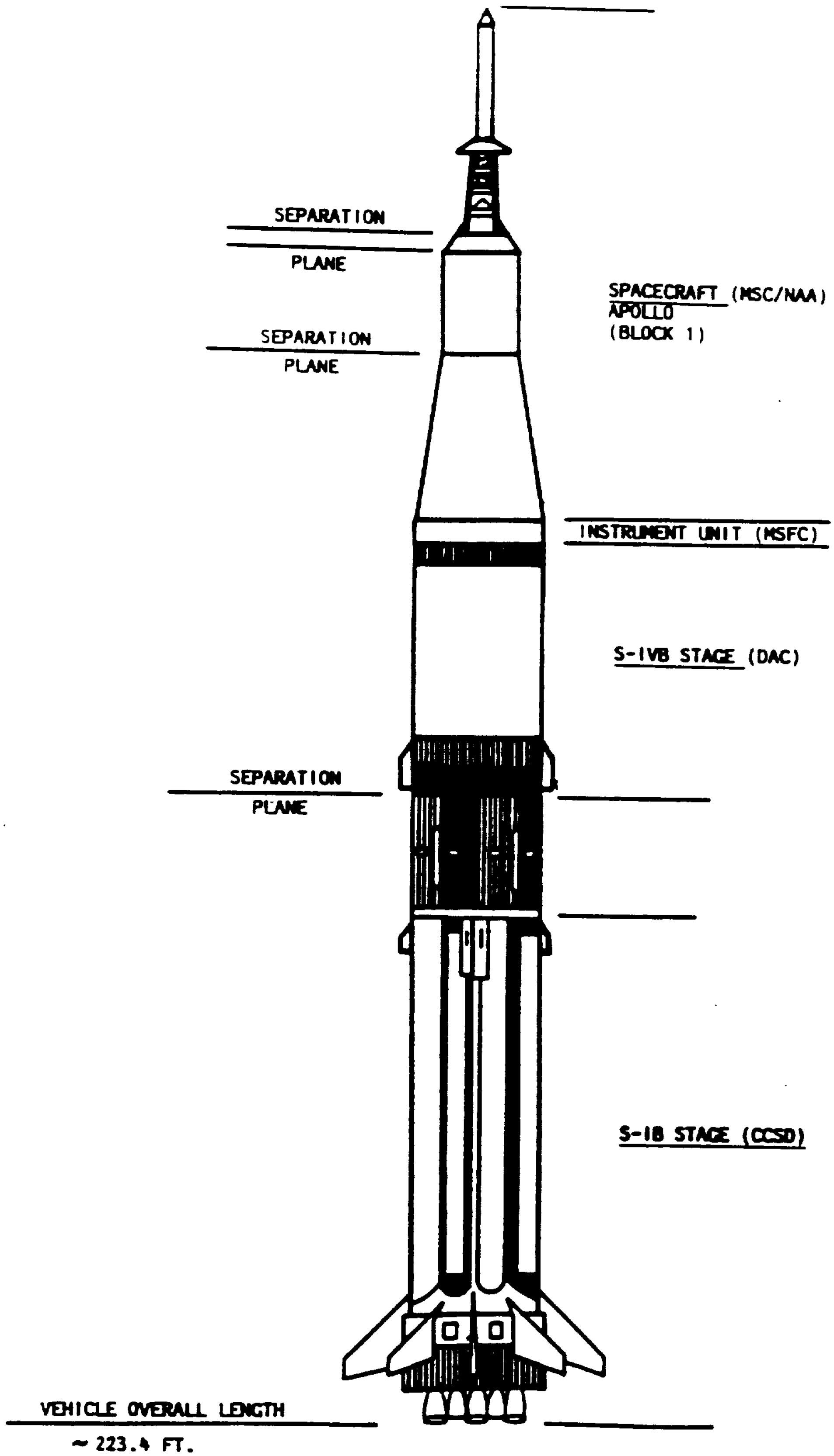
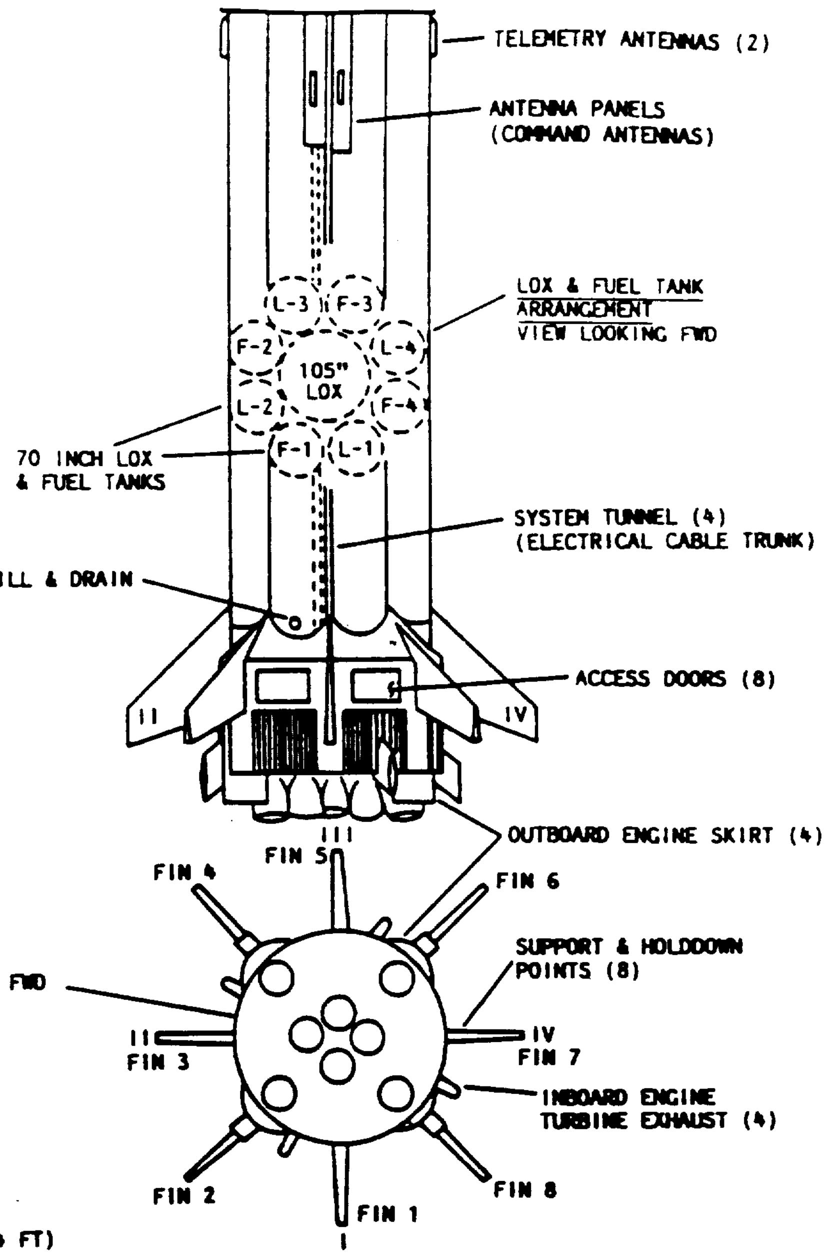


FIGURE 7-1. SATURN IB LAUNCH VEHICLE



DIAMETER 6.5 METERS (21.4 FT)
LENGTH 24.4 METERS (80.2 FT)
WEIGHT 452,240 KILOGRAMS (997,000 LBS) FUELED
38,347 KILOGRAMS (84,540 LBS) DRY
ENGINE H-1 (8)
PREPELLANTS: LIQUID OXYGEN 252,115 LITERS (66,609 GAL)
RP-1 (KEROSENE) 156,037 LETERS (41,225 GAL)
THRUST 7,294,700 NEWTONS (1,640,000 LBS)
CONTRACTOR: CHRYSLER CORPORATION

FIGURE 7-2. S-I/IB STAGE CONFIGURATION

later to 205,000 pounds. There was no ground test stage since the S-IB was so similar to the S-I that a separate ground test program was not necessary. The S-IB stage configuration is depicted in Figure 7-2.

S-IB stages were manufactured at the Michoud Assembly Facility by the stage contractor, Chrysler Corporation Space Division. After checkout, they were transported by barge to MSFC, where a number of systems tests were performed. This test series consisted of simulated flight tests, a propellant loading test, short duration firing (approximately 35 seconds), and a full duration firing (135-145 seconds). A total of 34 hot-fire tests were performed on the 12 stages, including four special combustion instability tests performed on stage S-IB-11.

Stage Propellant System. The stage propellant system (Figure 7-3) was composed of five liquid oxygen tanks, four RP-1 fuel tanks, propellant lines, control valves, vents, and pressurization subsystems (Figures 7-4 and 7-5). The sumps of each group of tanks were interconnected to provide uniform propellant levels and pressures. Loading of liquid oxygen and RP-1 fuel tanks was controlled by ground computers. After the RP-1 fuel had been loaded and just before the start of liquid oxygen loading, ground source nitrogen was bubbled through the RP-1 fuel suction lines to prevent temperature stratification. At the start of the automatic sequence, the RP-1 fuel tanks were pressurized with ground source nitrogen. During stage burn, fuel tank pressurization was maintained by helium from two 20-cubic foot spheres located above two of the fuel tanks. Ground source helium was bubbled through the liquid oxygen lines and tanks at the start of the automatic sequence to prevent temperature stratification in the engine liquid oxygen suction lines. Prior to engine ignition, the bubbling was discontinued and the liquid oxygen tanks were pressurized with helium from a ground source. After liftoff, the liquid oxygen tank pressurization (Figure 7-6) was maintained with gaseous oxygen from heat exchangers.

H-1 Engine Operation (Figure 7-6). A start signal ignited the solid propellant gas generator (SPGG) which accelerated the liquid oxygen and RP-1 fuel pumps. Increasing fuel pressure opened the main oxidizer valve which, in turn, opened the sequence valve permitting fuel pressure to rupture the hypergolic cartridge. Primary ignition occurred when the RP-1 fuel and hypergolic fluid contacted liquid oxygen in the thrust chamber. The injector fuel pressure opened the main RP-1 fuel valve and provided propellant flow to the liquid propellant gas generator (LPGG) which sustained turbine operation.

When the predetermined propellant level was sensed (within 1.8 seconds), the digital computer initiated inboard engine cutoff. Outboard engine cutoff was initiated by either the thrust OK switches, the backup timer (from the computer), or the fuel depletion probes. Both cutoff signals were routed through the stage switch selector. The cutoff signals opened the explosively actuated Conax valve equalizing the RP-1 fuel pressure at the main oxidizer valve. The valve closed to interrupt fuel flow and terminate engine operation.

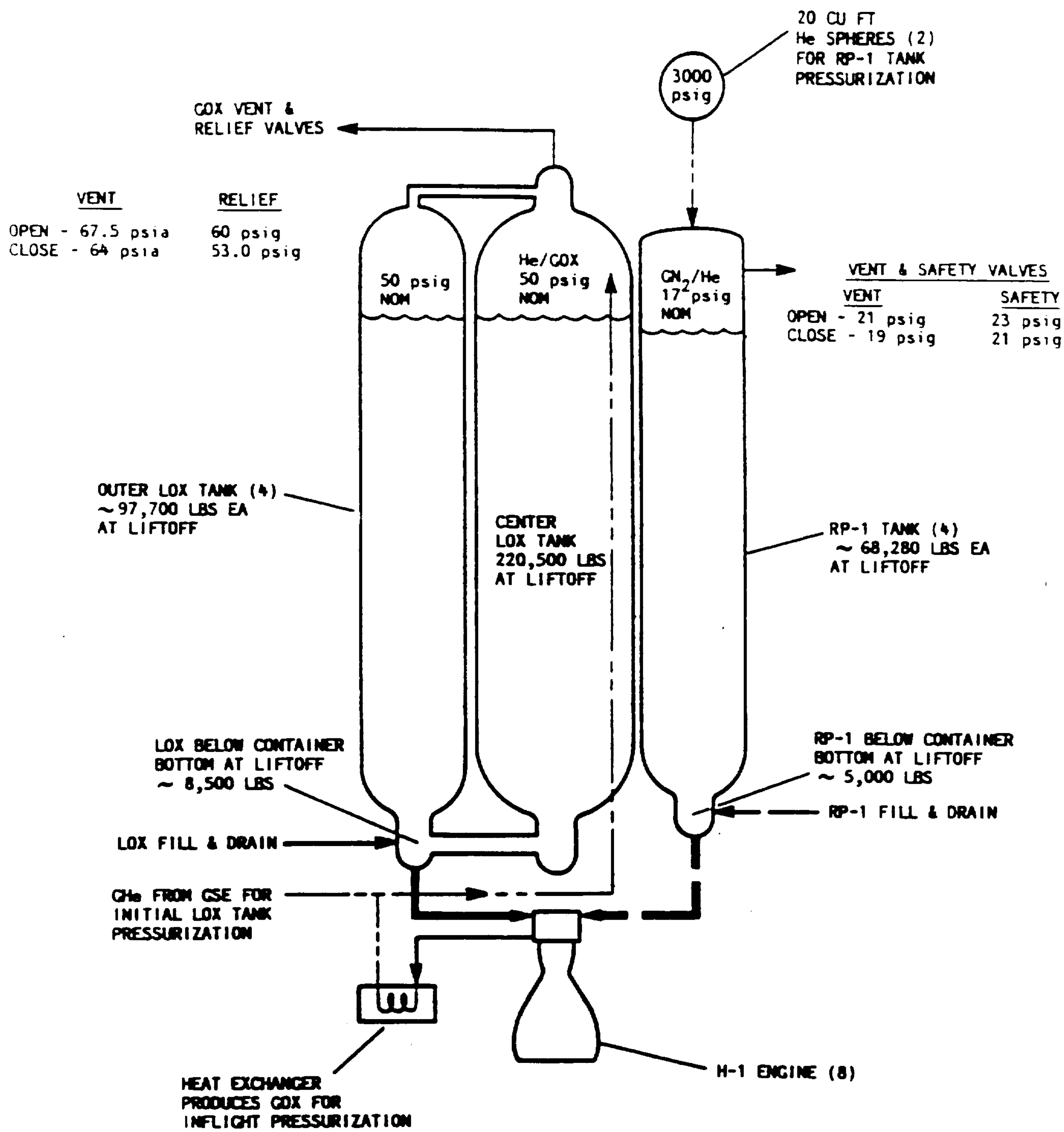


FIGURE 7-3. S-I/IB PROPELLANT SYSTEM

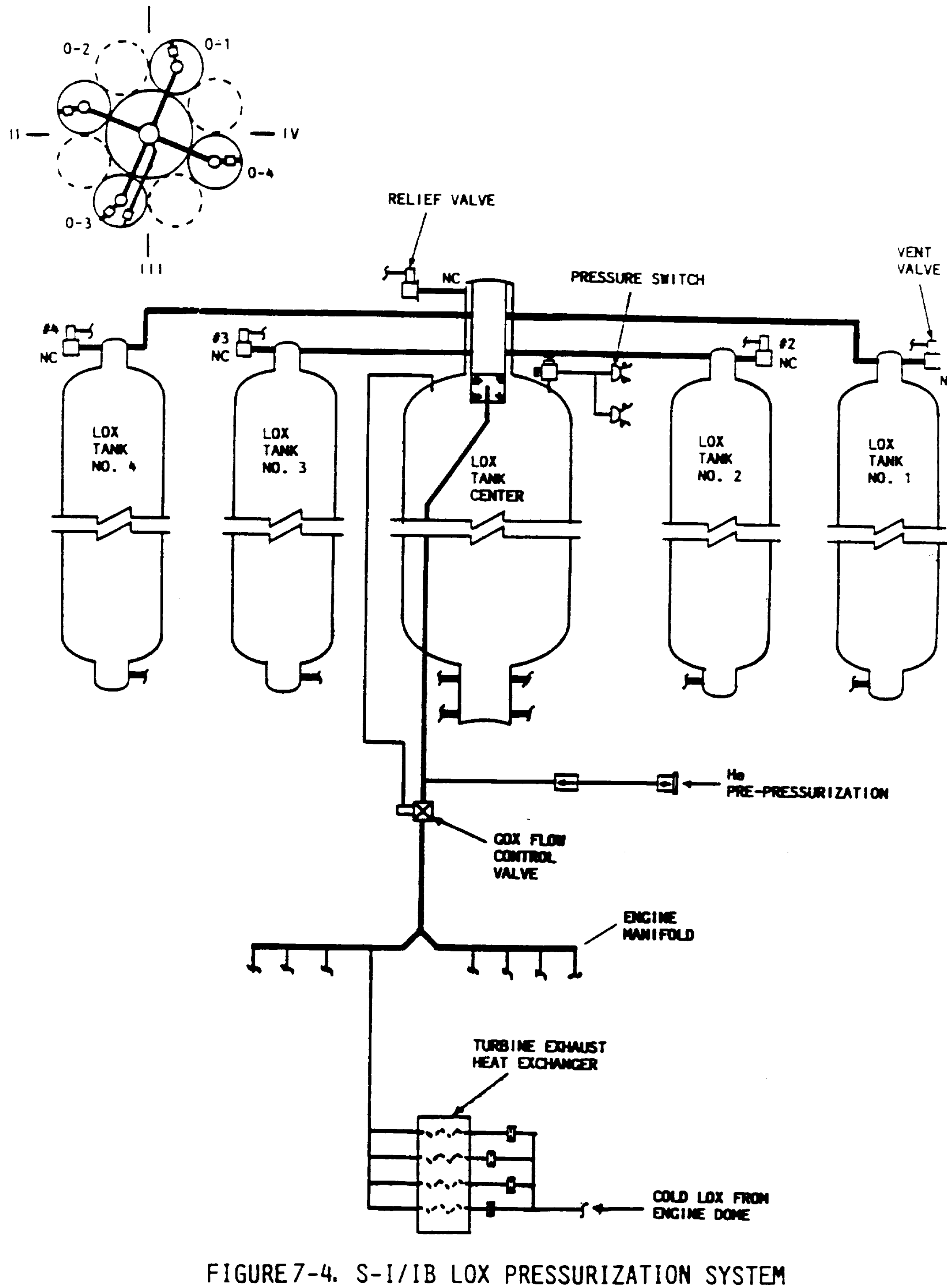


FIGURE 7-4. S-I/IB LOX PRESSURIZATION SYSTEM

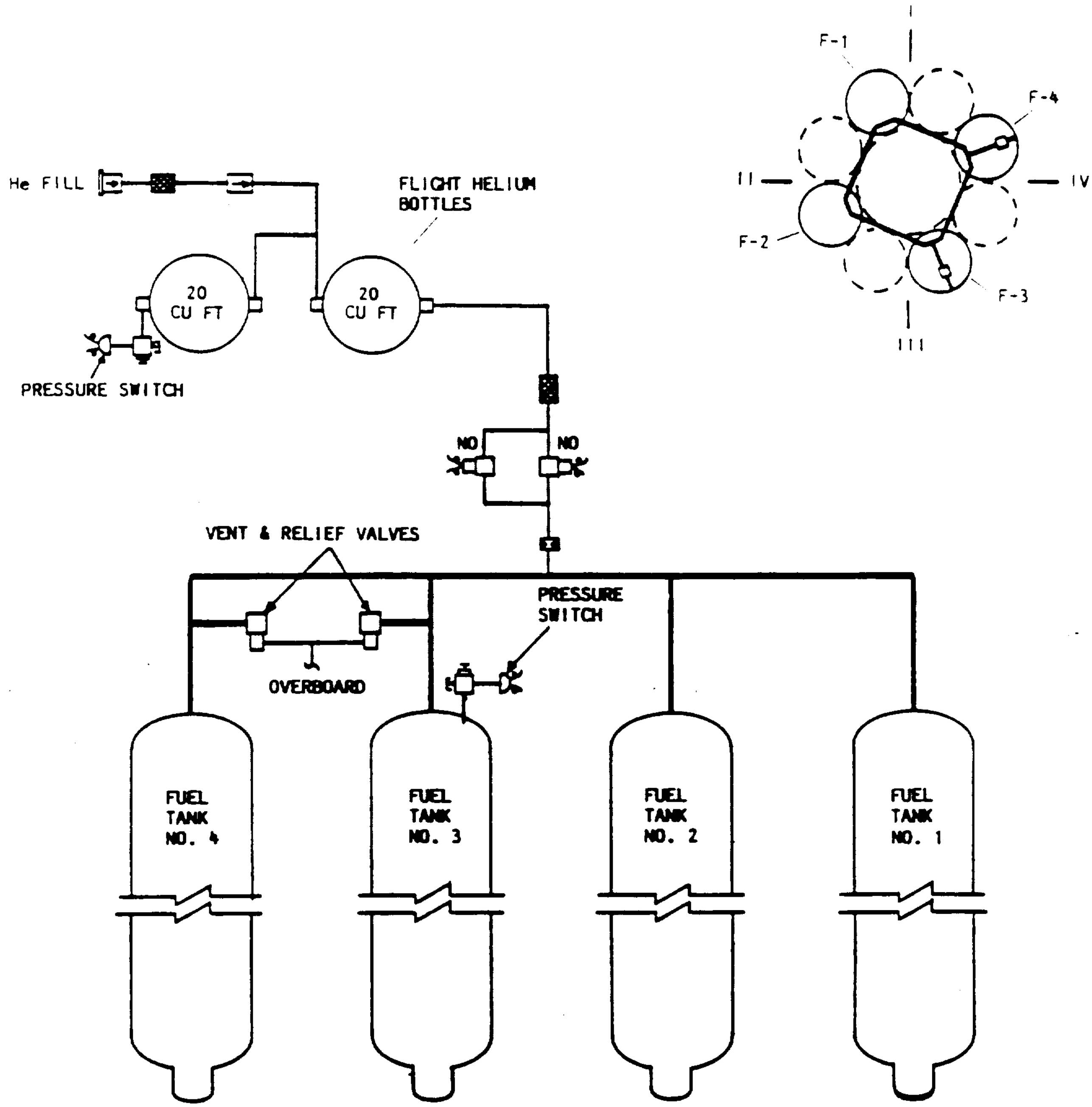


FIGURE 7-5. S-I/IB FUEL PRESSURIZATION SYSTEM

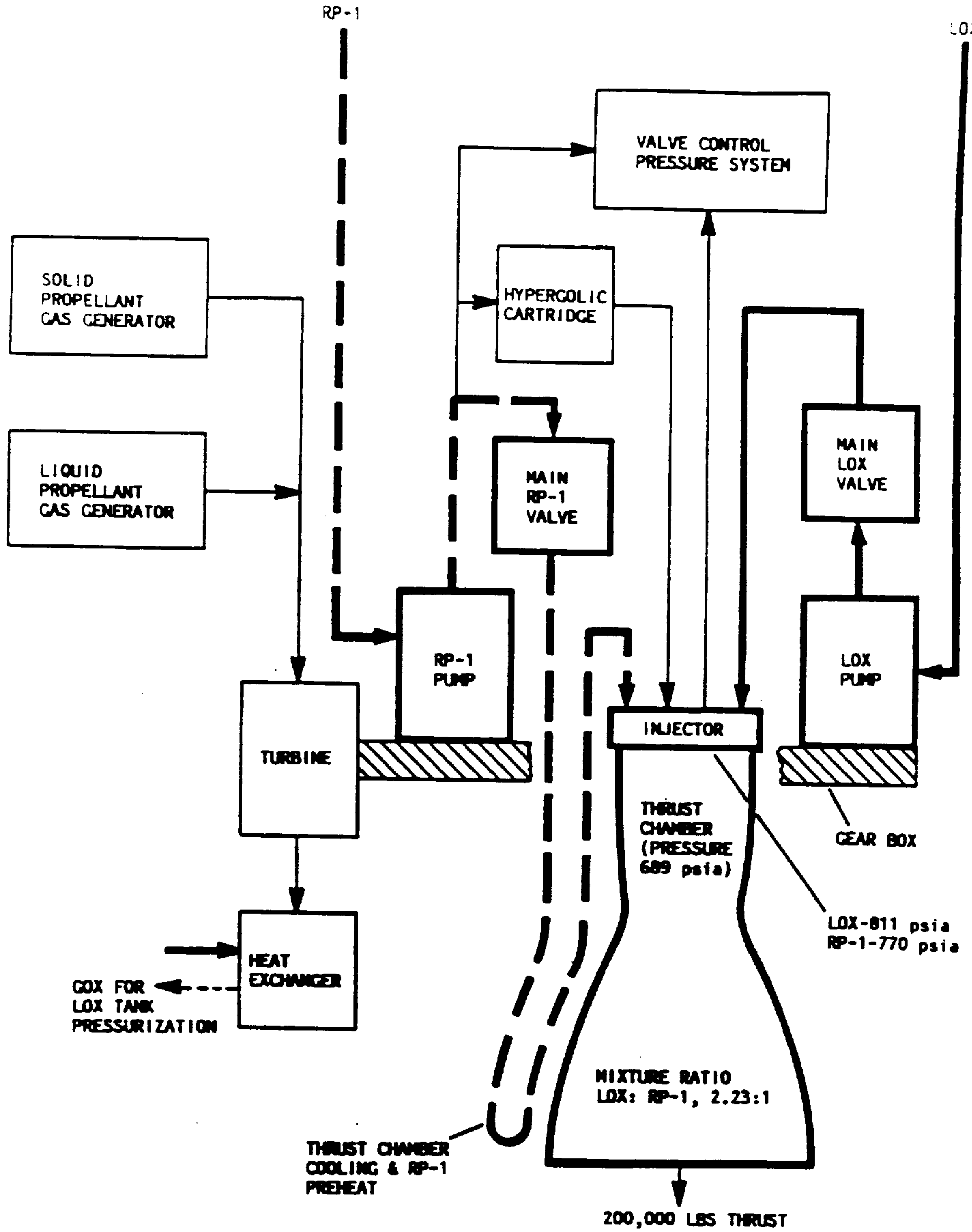


FIGURE 7-6. H-1 ENGINE SYSTEM

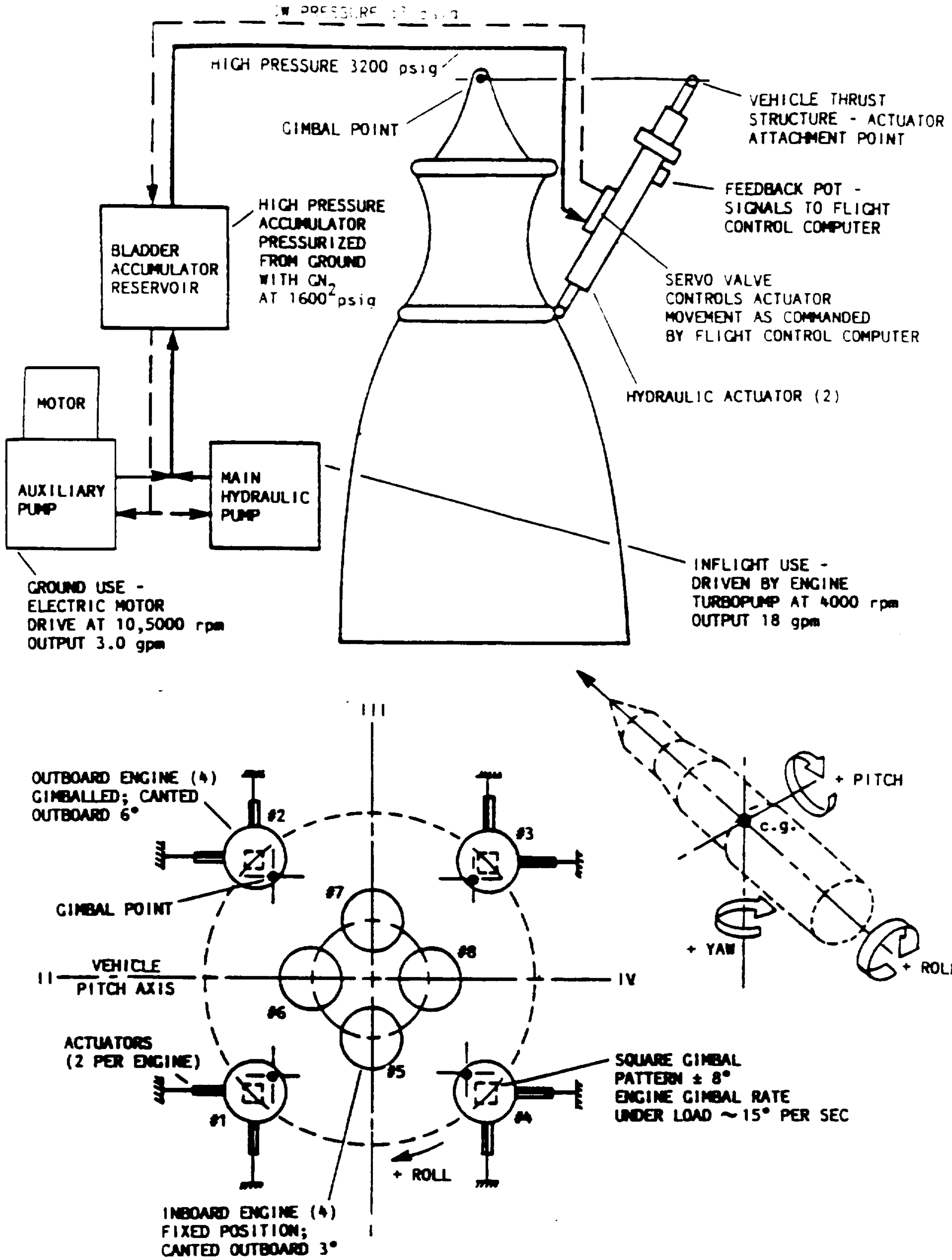


FIGURE 7-7. S-I/IB THRUST VECTOR CONTROL SYSTEM

Stage Thrust Vector Control System (Figure 7-7). Each of the four outboard H-1 engines was gimbal mounted on the stage thrust structure to provide engine thrust vectoring for vehicle attitude control and steering. Two hydraulic actuators were used to gimbal each engine in response to signals from the Flight Control Computer located in the Instrument Unit.

The actuators were part of an independent hydraulic system on each gimballed engine. Hydraulic fluid flowed to the actuators from the high pressure accumulator and returned to the low pressure reservoir. An electric motor driven auxiliary pump operated only during pre-launch checkout of the thrust vector control system.

Detail information on all aspects of the S-I and S-IB test and flight program is included in Table 7-1 and 7-2 of this appendix. The material is included both for information and historical purposes. Summary information on test program accomplishments includes appropriate information from this table and other data sources.

SATURN S-I/S-IB STAGES

S-I/S-IB Test Objectives

The S-I/S-IB test objectives were initially developed to demonstrate the feasibility of clustering a number of rocket engines and fuel and oxidizer tanks as a means of providing large thrust launch vehicles. Objectives were later added to demonstrate the capability for flight of the stage. Additional objectives were added as the test program evolved. Some of the major objectives were as follows:

- Evaluate H-1 engine performance in a clustered configuration
- Evaluate clustering of several tanks as flight storage containers for propellants
- Evaluate stage hydraulic system
- Evaluate stage thrust vector control system
- Evaluate performance of propellant fuel supply systems
- Evaluate performance of propellant pressurization systems
- Evaluate performance of aft compartment flame and heat shields
- Evaluate performance of propellant loading systems
- Evaluate performance of engine rough combustion cutoff system
- Evaluate capability of GSE to support vehicle launch operations.

Table 7-1. Saturn I Static and Flight Test Information from 03/28/60 through 12/20/60 (Sheet 1 Of 2)

(TI 1 & 1A)

TEST NO.	VEHICLE NO.	DATE	TIME (OF DAY)	DURATION (SEC)	SCHEDULED DURATION (SEC)	REASON FOR CUTOFF	NO. OF ENGINES	REMARKS	GIMBAL	HEAT EXCHGR	TAIL SHROUD	HEAT SHIELD	THROAT CURTAINS
SAT-01	SA-T	03/28/60	11:00 am	8	8	Scheduled Duration	2 (6&8)		NO	NO	NO	NO	NO
SAT-02	SA-T	04/06/60	11:15 am	6.9	8	Flashback	4 (Inbd)	Roll-out platform rolled under engines.	NO	NO	NO	NO	NO
SAT-03	SA-T	04/29/60	5:30 pm	8	8	Scheduled Duration	8		NO	NO	NO	NO	NO
SAT-04	SA-T	05/17/60	5:40 pm	25	25	Scheduled Duration	8	Kramer tail installed. Fiberglas board on lower star.	Engine No. 1	NO	NO	Heat Shld Installed ¼" steel	4 fiberglas curtains on outboard engines.
SAT-05	SA-T	05/26/60	5:04 pm	35	35	Scheduled Duration	8		Engine Nos 1&4	NO	NO	Heat Shld Installed ¼" steel	NO
SAT-06	SA-T	06/03/60	4:57 pm	75	75	Scheduled Duration	8		Engine Nos 1&4	NO	NO	Heat Shld Installed ¼" steel	NO
SAT-07	SA-T	06/08/60	5:00 pm	110	LOX depletion	High LOX pump inlet temps. pos. 4 and 5.	8	Nos. 4 & 5 LOX pump inlet temps rose to -284° F. Observer cutoff. First indication of LOX tank pressure collapse.	Engine Nos 1&4	NO	NO	Heat Shld Installed ¼" steel	New Curtain engine No. 3
SAT-08	SA-T	06/15/60	5:09 pm	121	LOX depletion	Low press. on water deflector	8	Deflector pressure dropped below redline. Cracked welds in expansion manifolds. Pressure decay LOX tank No. 4 had started.	Engine Nos 1&4	NO	NO	Heat Shld Installed ¼" steel	No new curtains.
SAT-09	SA-TI	12/02/60	6:01 pm	1.7	8	Rcc Eng. No. 2	8	Slow opening GG fuel valve due to low tail temps. Eroded first stage blades of all turbines. H-1014-cracked LOX dome	NO	2-Coil	Upper and lower shroud Installed X-258 on No. 4 bubble	Flight type heat shield X-258 and fiber-glas.	Steel throat plates all engines.
SAT-10	SA-TI	12/10/60	3:00 pm	6	6	Scheduled duration	2 (2&6)		NO	2-Coil			
SAT 11	SA-TI	12/20/60	4:40 pm	61	LOX depletion	Low L.P.T. pressure	8	Observer cutoff from L.P.T. pressure decay engines 4&5. LOX tank pres[illegible] decay tanks 1 &[illegible]	NO	2-Coil			

Table 7-1. Saturn I Static and Flight Test Information from 03/28/60 through 12/20/60 (Sheet 2 Of 2)

(TI.2 & 2A)

TEST NO.	BLAST CURTAINS	LOX SYTSEM PRESSURIZATION AND DEPLETION STUDY	ICE AND FROST SHIELDS	LOX HIGH PRESSURE BLEED LINE CONFIGURATION	TURBINE SPIN TYPE & COND TEMP & DUR	TURBINE SPINNER INITIATORS	GAS GENERATOR AUTO IGNITERS	HYPERGOL AND CONAX	BOATTAIL HEATING SYSTEM	MISCELLANOUS
SAT-01	1 Instl.	Ground pressurization thru 7" vent line	No	½-inch lines routed below flowmeter	Type A - Not conditioned	OA-AN9-2 2 megohms 28 volt	None	HYP-650439 CON-TEV-115-TX120	None	
SAT-02	4 Instl.	Ground pressurization thru 7" vent line	No	½-inch lines routed below flowmeter	Type A - Not conditioned	OAAI-AN9-2 50 megohms 28 volt	P/N 650900	HYP-650439 CON-TEV-115-TX120	None	
SAT-03	8 Instl.	Ground pressurization thru 7" vent line	No	½-inch lines routed overboard	Type A - Not conditioned	OAAI-AN0-2 50 megohms 28 volt	P/N 650900	HYP-650439 CON-TEV-115-TX120	None	
SAT-04	8 Instl.	Ground pressurization thru 7" vent line	No	Outboard-routed below flowmeter. Inboard - routed below prevalve.	Type A - Not conditioned	OAAI-AN9-2 50 megohms 28 volt	P/N 650900	HYP-650439 CON-TEV-115-TX120	None	
SAT-05	8 Instl.	Ground pressurization thru 7" vent line	No	Outboard-routed below flowmeter. Inboard - routed below prevalve.	Type A - Not conditioned	OAAI-AN9-2 50 megohms 28 volt	P/N 650900	HYP-650439 CON-TEV-115-TX120	None	Removed all 8 lube overboard check valves prior to the test.
SAT-06	8 Instl.	Ground pressurization thru 7" vent line	No	Outboard-routed below flowmeter. Inboard - routed below prevalve.	Type A - Not conditioned	OAAI-AN9-2 50 megohms 28 volt	P/N 650900	HYP-650439 CON-TEV-115-TX120	None	Modified LOX and fuel pressurizing circuit to prevent rapid cycling early in test. No lube overboard check valves.
SAT-07	8 Instl.	Pressure decay LOX tank No. 1. Lost baffle on 7" vent and 4" pressure line. Ground pressurization thru 7" vent line	No	Outboard also routed below prevalve.	Type A - Not conditioned	OA-AN9-2 2 megohms 28 volt	P/N 650900 except pos. 1 had 1 S-30 GG igniter used as auto.	HYP-650439 CON-TEV-115-TX120	None	No lube overboard check valves.
SAT-8	8 Instl.	Pressurization through "missile whistle", installed at GOX inlet.	No	Outboard also routed below prevalve.	Type A - Not conditioned	OAAI-AN9-2 50 megohms 28 volt	P/N 650900	HYP-650439 CON-TEV-115-TX120	None	No lube overboard check valves.
SAT-9	8 Instl.	Open 30-inch standpipes. GOX pressurization. "Minnow Basket" installed.	Ice shlds installed test design	All engines - ½-inch lines routed below flowmeter	Type A-1 78°F for 48 hrs. except pos. 4 60°F	OAAI-AN9-2 50 megohms 28 volt	P/N 650900	HYP-650818 CON-TEV-115-TX120	4 electric blower heaters. 57,000 BTU/HR at 825 cfm each	2 teflon-lined flexhose, 3/4-inch, P/N 10413270B, installed in fuel sphere system, replacing the all-metal flexhoses. Removed the inboard engine lube overboard check valves.
SAT-10	8 Instl.	Open 30-inch standpipes. GOX pressurization. "Minnow Basket" installed.	Ice shlds installed test design	All engines - ½-inch lines routed below flowmeter	Type A-1 55°F for 24 hrs.	OAAI-AN9-2 50 megohms 28 volt	P/N 650900	HYP-650818 CON-TEV-115-TX120	4 Herman-Nelson BT 400-10 gasoline burners with electric blowers. 400,000 BUT/HR each.	Fuel spheres pressure switch put into "prep complete" circuit and made to control the spheres pressurizing valves.
SAT-11	8 Instl.	Standpipes in tanks No. 2 & 4 severely damaged.	Frst shld installed between F-2 and 0-3	All engines- ½-inch lines routed below flowmeter	Type A-1 75°F for 24 hrs.	OAAI-AN9-2 50 megohms 28 volt	P/N 650900	HYP-650818 CON-TEV-115-TX120	Ditto heaters. Eng. compartment fed by 18" ring manifold with 24 6" hoses attchd.	Inboard fuel lube overboard restrictor check valves installed.

Table 7-1. Saturn I Static and Flight Test Information from 01/31/61 through 02/14/61 (Sheet 1 Of 2)

(TI.1B)

TEST NO.	VEHICLE NO.	DATE	TIME (OF DAY)	DURATION (SEC)	SCHEDULED DURATION (SEC)	REASON FOR CUTOFF	NO. OF ENGINES	REMARKS	GIMBAL	HEAT EXCHGR	TAIL SHROUD	HEAT SHIELD	THROAT CURTAINS
SAT-12	SA-T1	01/31/61	4:48 pm	113	LOX depletion	Scheduled duration	8	Cutoff from thrust OK pressure switch engine No. 3.	Engine Nos 1&4	2-Coil			Engine Nos. 1&4, fiberglas and Refrasil. No. 4 had ½ area coated with PR-1910.
SAT-13	SA-T1	02/14/61	4:48 pm	109	LOX depletion	Scheduled duration	8	Cutoff from thrust OK pressure switch engine No. 3.	Engine Nos. 1, 3, & 4	2-Coil		One panel had D-100 coating.	All outboard engines fiberglas coated with PR-1910.

Table 7-1. Saturn I Static and Flight Test Information from 01/31/61 through 02/14/61 (Sheet 2 Of 2)

(TI.2B)

TEST NO.	BLAST CURTAINS	LOX SYSTEM PRESSURIZATION AND DEPLETION STUDY	ICE AND FROST SHIELDS	LOX HIGH PRESSURE BLEED LINE CONFIGURATION	TURBINE SPIN TYPE & COND TEMP & DUR	TURBINE SPINNER INITIATORS	GAS GENERATOR AUTO IGNITERS	HYPERGOL AND CONAX	BOATTAIL HEATING SYSTEM	MISCELLANOUS
SAT-12	8 Instl.	"Chinese Hats" installed. Film coverage inside 70" tanks.	All frost shlds in excpt on each side of F-4	All engines-½-inch lines routed below flowmeter	Type A-1 70°F for 72 hrs.	Microloc 21085 500 volt	P/N 650900	HYP-650818 CON-TEV-115-TX120		Removed 600 and installed 700± 7 psia thrust OK pressure switches. SA-1 type suction line installed at Position No. 1
SAT-13	8 Instl.	"Chinese Hats" installed. Film coverage inside 70" tanks.	All frost shlds in excpt on each side of F-4	All engines-½-inch lines routed below flowmeter	Type A-1 70°F for 72 hrs.	McCormick-Selph D 803400, 500 volt	P/N 650900	HYP-650818 CON-TEV-115-TX120		A 0.125-inch diameter orifice installed in 1500-3000 psig fuel spheres supply line. SA-1 pressurizing sequence with flight simulation of fuel tank pressure switch in vacuum chamber

Table 7-1. Saturn I Static and Flight Test Information from 04/29/61 through 07/18/61 (Sheet 1 Of 2)

(TI.3 & 3A)

TEST NO.	VEHICLE NO.	DATE	TIME (OF DAY)	DURATION (SEC)	SCHEDULED DURATION (SEC)	REASON FOR CUTOFF	NO. OF ENGINES	REMARKS	GIMBAL	HEAT EXCHGR	TAIL SHROUD	HEAT SHIELD	THROAT CURTAINS
SA-01	SA-1	04/29/61	4:39 pm	30	30	Scheduled duration	8	LOX bubbling - 16 secs.	Eng. 1, 2, 3 and 4	2-Coil	Bubble covers No.3 ctd with D-100. No.4 ctd with X-258. No. 1 & 2 plain covers.	H.S. insulated with D-100. and X-258	All outboard curtains were flight rejects.
SA-02	SA-1	05/05/61	4:12 pm	44	LOX depletion	Tail fire	8	Automatic cutoff by engs 1&2 fire detection harness. Ruptured SPGG pressure transducer. LOX bubbling - 16 secs.	Eng. 1, 2, 3 and 4	2-Coil		D-100, X-258 Alum. Tape and green fiberglas insulat.	All outboard curtains were flight rejects.
SA-03	SA-1	05/11/61	3:48 pm	111	LOX depletion	Scheduled duration. Low LOX level in tank 0-2	8	Cutoff intended from low level switch in LOX tank 2 or 4. Did not employ flight cutoff sequencing.	Eng. 1, 2, 3 and 4	2-Coil			Ditto with new curtains on engs No. 3 and 4.
SAT-14	SA-T2	06/27/61	3:52 pm	30	30	Scheduled duration	8	LOX bubbling - 6 secs. Engines 1 & 2 had used PR-1910 coated curtains for SAT-14 through SAT-19.	No. 4 only servo valve P/N 16-X119	2-Coil	Nos. 1,2 bubbles uncovered No.3 plain cov. No.4 D-100 on bubble	X-258 @ Eng. Nos. 1 & 2 D-100 rest of H.S.	Eng. Nos. 1 & 2 - see remarks. Eng. Nos. 3 & 4 - used fiberglas coated with PR-1910.
SAT-15	SA-T2	07/07/61	4:20 pm	118	LOX depletion	Observer cutoff close to scheduled duration.	8	Scheduled for flt seq. cutoff. Cutoff by observer. LOX depletion circuit improperly set up. LOX bubbling - 16 seconds.	Eng. 3 & 4	2-Coil	Nos. 1&2 bubbles uncovered Nos. 3&4 bubble covers ctd with X-258	X-258 at Eng. Nos. 3&4.D-100 on rest of H.S. except PR-1910 on panel 3 @ fin IV	No. 4 - PR-1910 coated with silicone rubber no. 3 - Normco 4551 stripped with silicone.
SAT-16	SA-T2	07/18/61	2:41 pm	116	LOX depletion	Secheduled duration	8	Flt cutoff seq. initiated by switch in No. 3 LOX tank. Time: 110.47 and 116.16 sec.	Eng. 1, 3, and 4.	2-Coil		Same except panel 3 @ fin IV coated with X-258	Nos. 3 & 4 - reverse of SAT-15

Table 7-1. Saturn I Static and Flight Test Information from 04/29/61 through 07/18/61 (Sheet 2 Of 2)

(TI.4 & 4A)

TEST NO.	BLAST CURTAINS	LOX SYSTEM PRESSURIZATION STUDY	ICE AND FROST SHIELDS	LOX HIGH PRESSURE BLEED LINE CONFIGURATION	TURBINE SPIN TYPE & COND TEMP & DUR	TURBINE SPINNER INITIATORS	GAS GENERATOR AUTO IGNITERS	HYPERGOL AND CONAX	BOATTAIL HEATING SYSTEM	MISCELLANOUS
SA-01	No	GN2 pre-pressurization. 30" standpipes with "Chinese Hats"	Ice shld only. No frst shld	½" lines routed below flowmeter.	Type A-1 60°F for 72 hrs.	Fleming F61001-100, 500 volt	P/N 650900	HYP-650818 CON-1804001-01 300°F No fire	Launch type compressor & after-cooler. Heat delivered to eng comp. thru water quench nozzles.	Canister cooling package checkout - SA-01 through SA-03. Simulated short cable mast installed at fin IV - SA-01 through SA-03.
SA-02	No	GN2 pre-pressurization. 30" standpipes with "Chines Hats"	Ice shld only. No frst shld	½" lines routed below flowmeter.	Type A-1 60°F for 72 hrs.	Fleming F61001-100, 500 volt	P/N 650900	HYP-650818 CON-1804001-01 300°F No fire	New motor compressor installed replacing unit which malfunctioned during SA-01	Launch countdown (2 day) SA-02 and SA-03
SA-03	No	GN2 pre-pressurization. 30" standpipes with "Chinese Hats"	Ice shld only. No frst shld	½" lines routed below flowmeter.	Type A-1 60°F for 72 hrs.	McCormick-Selph D803400 500 volt	P/N 650900	HYP650818 CON-1804001-01 300°F No fire	New motor compressor installed replacing unit which malfunctioned during SA-01	
SAT-14	No	SA-2 flt type standpipes. GN2 pre-pressurization.	Ice & frst shlds as for SAT-13	½" lines routed below flowmeter	Type A-1 60°F for 72 hrs.	Fleming F61001-100, 500 volt	P/N 650900	HYP-650818 CON-1804001-01 300°F No fire	Ditto SA-02 and SA-03	Position indicators on GG control valves. Structure reinforcement, strut and brace installed at eng. position No. 4 for "damping factor" study.
SAT-15	No	SA-2 flt type standpipes. GN2 pre-pressurization.	Ice & frst shlds as for SAT-13	"Hard Priming" investigation. See Test Request	Type A-1 45°F for 72 hrs.	Fleming F61001-100, 500 volt	P/N 650900	HYP-650818 CON-1804001-01 300°F No fire	Ditto SA-02 and SA-03	Strut & brace-positions 3 & 4; hydraulic servo valves, P/N 16-X119, on position 4 and P/N 22157 on pos. 3 for damping factor study. Also "twang" test @ pos. 2. Test of flt version of TV camera and light in engine compartment.
SAT-16	No	Helium pre-pressurization	Ice & frst shlds for SAT-13	Ditto, except pos Nos. 5 & 6 line size increased to 3/4 inch.	Type A-1 45°F for 72 hrs.	McCormick-Selph D 803400 500 volt	P/N 650900	HYP-650818 CON-1804001-01 300°F No fire	Ditto - SA-02 & SA-03	"Twang" test - pos. 2, struts and braces - pos. 1, 3, and 4. Protective cvrs around eng throats. TV camera and light in engine compartment.

Table 7-1. Saturn I Static and Flight Test Information from 08/03/61 through 08/25/61 (Sheet 1 Of 2)

(TI.3B)

TEST NO.	VEHICLE NO.	DATE	TIME (OF DAY)	DURATION (SEC)	SCHEDULED DURATION (SEC)	REASON FOR CUTOFF	NO. OF ENGINES	REMARKS	GIMBAL	HEAT EXCHANGE	TAIL SHROUD	HEAT SHIELD	THROAT CURTAINS
SAT-17	SA-T2	08/03/61	5:09 pm	1.3	LOX depletion	High L.P.I. temperature	8	Observer cutoff - No. 1 L.P.I. temp indication went full scale. Brkn wire on temp. transducer	No	2-Coil	Same except new X-258 on No. 4 cov	New X-258 @ eng. Nos. 3&4. Rest same.	No. 3 - Fiberglas coated with PR-1910 and outer layer of refrasil. No. 4 - 2 layer fiberglas and outer layer refrasil.
SAT-18	SA-T2	08/07/61	3:40 pm	123	LOX depletion	Scheduled duration	8	Flt cutoff seq. Switch in No. 2 LOX tank. Time: 118.21 and 123.80 sec.	Eng. 1, 2, 3, and 4.	2-Coil	No new coating.	Ditto (No damage during SAT-17)	
SAT-19	SA-T2	08/25/61	3:42 pm	114	Approx. 115 sec.	Secheduled duration	8	"Eng. Out" test. No. 5 eng. cut off manually at a predetermined LOX lev. 20 secs. before end of test.	Moog vlvs P/N 16-119 eng Nos. 1 2,3 &4	5 2-Coil, 4-Coil on eng. Nos. 6, 7, and 8	New X-258 Nos. 3&4 bubble covers.	Several panels recoated with X-258	Nos. 3 and 4 - two layers of fiberglas coated with PR-1910.

Table 7-1. Saturn I Static and Flight Test Information from 08/03/61 through 08/25/61 (Sheet 2 Of 2)

(TI.4B)

TEST NO.	BLAST CURTAINS	LOX SYSTEM PRESSURIZATION STUDY	ICE AND FROST SHIELDS	LOX HIGH PRESSURE BLEED LINE CONFIGURATION	TURBINE SPIN TYPE & COND TEMP & DUR	TURBINE SPINNER INITIATORS	GAS GENERATOR AUTO IGNITERS	HYPERGOL AND CONAX	BOATTAIL HEATING SYSTEM	MISCELLANOUS
SAT-17	No	GN2 pre-pressurization	Ice & frst shlds for SAT-13	Ditto, except pos Nos. 5 & 6 line size increased to 3/4 inch.	Type A-1 45°F for 72 hrs.	McCormick-Selph D 803400 500 volt	P/N 650900	HYP-650818 CON-1804001-01 300°F No fire	Ditto - SA-02 & SA-03	Helium bubbling and L.P. I. temp measurement re-oriented to comply with SA-1 configuration. Struts and braces pos. 1, 2, 3 and 4. Vibration test of panels mtd in No. 3 fuel tank forward skirt. Color motion camera in eng compartment; special meas. inside No. 7 exhaust duct; temp meas. lower end of No. 5 eng.
SAT-18	No	GN2 pre-pressurization	Ice & frst shld as for SAT-13	Ditto, except pos Nos. 5 & 6 line size increased to 3/4 inch.	Type A-1 45°F for 72 hrs.	McCormick-Selph D 803400 500 volt	P/N 650900	HYP-650818 CON-1804001-01 300°F No fire	Ditto - SA02 and SA-03	Ditto SAT-17, plus hydraulic oil temp. stdy with auxiliary pump eng. No. 1 turned on at X-30 minutes.
SAT-19	No	GN2 pre-pressurization	Ice & frst shld as for SAT-13	Ditto, except pos Nos. 5 & 6 line size increased to 3/4 inch.	Type A-1 45°F for 72 hrs.	McCormick-Selph D 803400 500 volt	P/N 651139	HYP-650818 CON-1804001-01 300°F No fire	Ditto - SA02 and SA-03	X-258 covers on engine No. 4 aspirator and reinforcement rings.

Table 7-1. Saturn I Static and Flight Test Information from 10/10/61 through 02/06/62 (Sheet 1 Of 2)

(TI.5 &5A)

TEST NO.	VEHICLE NO.	DATE	TIME (OF DAY)	DURATION (SEC)	SCHEDULED DURATION (SEC)	REASON FOR CUT OFF	ACCUMUL. VEHICLE SAT ONLY	REMARKS	GIMBAL	HEAT EXCHGR	TAIL SHROUD	HEAT SHIELD	THROAT CURTAINS
SA-04	SA-2	10/10/61	5:00 pm	32	30	Scheduled duration		Eng. No. 1 - PR-1910 on fiberglas; Eng. No.2 4551 on fiberglas (SA-04 & SA-05. Helium bubbling - 30 secs.	Eng. 1, 2,3,& 4	4-Coil	Shroud bubbles ctd with X-258. No covers	X-258 & D-100	No. 3 - 4551 on fiberglas on re-frasil. No. 4 4551 on fiberglas with silicone rubber. Nos. 1&2 see remarks.
SA-05	SA-2	10/24/61	4:41 pm	119	LOX depletion	Scheduled duration		Flt. cutoff seq. initiated by switch in No. 4 LOX tank. Time: 112.9 and 118.9 sec.	No	4-Coil	No new coating	X-258 & D-100	Nos. 3 & 4 - 4551 on fiberglas with aluminum fiberglas on refrasil.
SAT-20	SA-T3	11/30/61	5:01 pm	95	LOX depletion	Automatic- No. 7 fire harness	1277 Sec.	All throat curtains (SAT-20 thru SAT-24) were Normco 4551 on 2 layers of fiberglas with loose outer layer of aluminized fiberglas on refrasil.	Eng. # 3 only	All 4-coil except Eng. #2 has 2-coil	No insulation. No bubble covers.	All X-258 except D-100 on fin II, panel Nos 3, 4, & 5 fin III, panel No. 2	See remarks for type. Eng. #1 - S/N 22, #2 - S/N 20, #3 - S/N 26, #4 - S/N 34
SAT-21	SA-T3	12/19/61	3:42 pm	68	LOX depletion	Observer - High No. 1 bearing temp indication	1345 Sec.	Erroneous No. 1 bearing temp. indication due to broken thermocouple wire (harness).	Eng. 1, 2, & 4	All 4-coil except Eng. #2 has 2-coil	No insulation. No bubble covers.	Ditto, except new X-258 on fin IV panel 4 & access covers at Pos. 1, 3 and 4	Eng. #1 - S/N 22, Eng. #2 - S/N 20, #3 - S/N 28, #4 S/N 34
SAT-22	SA-T3	01/18/62	4:47 pm	122	LOX depletion	Scheduled duration	1467 Sec.	"Eng. Out" test: Eng. 5 cut off @ 104.54 sec. Inboard cutoff @ 115.92 Outboard @ 121.90 (Flt cutoff sequence).	Eng. 1, 2, 3, & 4	All 4-coil except Eng. #2 has 2-coil	No insulation. No bubble covers	All X-258 except D-100 on fin II panels 4 & 5 and fin III, panels 2 & 3	Eng. #1 - S/N 22, Eng. #2 - S/N 20, #3 - S/N 28, #4 S/N 34
SAT-23	SA-T3	02/06/62	4:40 pm	46	Approx. 50 sec.	Observer- Completion of test program	1513 Sec.		Eng. 1, 2, 3 & 4	All 4-coil except Eng. #2 has 2-coil	No insulation. No bubble covers	Ditto SAT-20except panel 4, fin IV had R&D insulation - potassium titanate & silica	Eng. #1 - S/N 22, #2 - S/N 20, #3 - S/N 28, #4 - S/N 48.

Table 7-1. Saturn I Static and Flight Test Information from 10/10/61 through 02/06/62 (Sheet 2 Of 2)

(TI.6 & 6A)

TEST NO.	GG CONTROL VALVE	LOX SYSTEM PRESSURIZATION STUDY	LOX HIGH PRESSURE BLEED LINE CONFIGURATION	RATE GYRO LOCATION STUDY	TURBINE SPIN TYPE & COND TEMP & DUR	TURBINE SPINNER INITIATORS	GAS GENERATOR AUTO IGNITERS	HYPERGOL AND CONAX	MISCELLANOUS
SA-04	Cracking check valves	GN2 pre-pressurization	½-inch lines routed below flowmeter	No	Type - A-I 40°F for 72 hrs.	Fleming F61001-100, 500 volt	P/N 651139	HYP-650818 CON-1804001-01 300°F - No fire	Boattail heating system for SA-04 and SA-05 - same as for SAT-14 through SAT-19.
SA-05	Cracking check valves	Helium pre-pressurization	½-inch lines routed below flowmeter	(See Misc)	Type - A-I 40°F for 72 hrs.	Fleming F61001-100, 500 volt	P/N 651139	HYP-650818 CON-1804001-01 300°F - No fire	Hyd. pump surface temp. measurements. Heat shield test panel with thermocouples. Rate gyro environmental study on spider beam - fin IV.
SAT-20	Yoke valves except #6 & 8 have cracking valves.	Helium pre-pressurization	½-inch lines routed below flowmeter	No	Type A-I 40°F for 48 hrs.	Fleming F61001-100, 500 volt	651139	HYP-650818 CON-1804001-01	Type 17-121F hyd. actuators installed pos. No. 3. "Saturn Vehicle Boattail Conditioning System" (packaged unit) used for tests SAT-20 through SAT-24. Frost shield removed.
SAT-21	Yoke valves except #6 & 8 have cracking valves.	Helium pre-pressurization	½-inch lines routed below flowmeter	Rate gyro pkg mounted at sta. 145 for performance eval. & vibration and temp. study.	Type A-I 50°F for 48 hrs.	McCormick-Selph D-803400, 500 volt	651139	HYP-650818 CON-1804001-01	
SAT-22	Yoke valves except #6 & 8 have cracking valves.	Helium pre-pressurization	½-inch lines routed below flowmeter	Rate gyro pkg mounted at sta. 145 for performance eval. & vibration and temp. study.	Type A-I 50°F for 72 hrs.	Fleming F61001-100, 500 volt	651139	HYP-650818 CON-1804001-01	Test of R&D radiation detection equipment in eng. compartment. Fuel and LOX tank liquid level investigation in conjunction with "Eng. Out" test. Hyd. actuator type 17-121F with Hyd. Research servo valves pos. #3. Servo valves (P&Y) pos. #4 mtd. on blocks with drilled bypass between control ports. The chk valve removed from No. 5 fuel lube overboard line.
SAT-23	Yoke valves all engines	GN2 pre-pressurization	Ditto, except eng. No. 3 bleed line simulating SA-5 configuration.	Rage gyro pkg. mounted at sta. 145 for performance eval. & vibration and temp. study.	Type A-I 50°F for 72 hrs.	Fleming F61001-100, 500 volt	651139	HYP-650818 CON-1804001-01	LOX system study with 7-inch vent & No. 1 relief valve opening during test. Test of R&D radiation detection equipment in engine compartment. Hyd. data - same as per test SAT-22.

Table 7-1. Saturn I Static and Flight Test Information 02/20/62 (Sheet 1 Of 2)

(TI.5B)

TEST NO.	VEHICLE NO.	DATE	TIME (OF DAY)	DURATION (SEC)	SCHEDULED DURATION (SEC)	REASON FOR CUT OFF	ACCUMUL. VEHICLE SAT ONLY	REMARKS	GIMBAL	HEAT EXCHGR	TAIL SHROUD	HEAT SHIELD	THROAT CURTAINS
SAT-24	SA-T3	02/20/62	4:38 pm	55	LOX depletion	Fire detection harness	1568 Sec.	Observer cutoff but No. 6 fuel and Nos. 1 & 5 LOX harnesses gave cut-off signals only milli-seconds later.	Eng. 1 and 4	Pos. No. 3 LOX-to -HE line discon-nected	No insul-ation. No bubble covers	Ditto SAT -23except a new R&D panel on fin IV, panel 4.	Eng. #1 - S/N 22, #2 - S/N 20, #3 - S/N 28, #4 - S/N 48.

Table 7-1. Saturn I Static and Flight Test Information
02/20/62 (Sheet 2 Of 2)

(TI.6B)

TEST NO.	GG CONTROL VALVE	LOX SYSTEM PRESSURIZATION STUDY	LOX HIGH PRESSURE BLEED LINE CONFIGURATION	RATE GYRO LOCATION STUDY	TURBINE SPIN TYPE & COND TEMP & DUR	TURBINE SPINNER INITIATORS	GAS GENERATOR AUTO IGNITERS	HYPERGOL AND CONAX	MISCELLANOUS
SAT-24	Yoke valves all engines	GN2 pre-pressurization	Ditto, except eng. No. 3 bleed line simulating SA-5 configuration.	Rage gyro pkg. mounted at sta. 145 for performance eval. & vibration and temp. study.	Type A-I from 8 different lots, 50°F for 72 hrs.	Fleming F61001-100, 500 volt	651139	HYP-650818 CON-1804001-01	Welded SPGG case on eng. No. 5 High relief and vent switch settings on LOX tank vent valves to prevent venting during test. Study of GOX in LOX tanks at end of test. New Texas Instrument Co.all-metal microswitches, P/N AT-25-1-2, installed in 7-inch LOX vent valve and Nos. 2 & 4 LOX prevalves. Hyd. data - same as per test SAT-22 and 23.

Table 7-1. Saturn I Static and Flight Test Information from 04/10/62 through 07/17/62 (Sheet 1 Of 2)

(TI.7 & 7A)

TEST NO.	VEHICLE NO.	DATE	TIME (OF DAY)	DURATION (SEC)	SCHEDULED DURATION (SEC)	REASON FOR CUTOFF	ACCUMUL. VHCL TIME SAT ONLY	REMARKS	GIMBAL	HEAT EXCHANGE	TAIL SHROUD	HEAT SHIELD	THROAT CURTAINS
SA-06	S-I-3	04/10/62	5:13 pm	31	30	Scheduled duration completed		Cutoff given after completion of gimbal program.	Eng. 1 2, 3, & 4	4-Coil	No shroud bubble cvrs inside shrd ctd with X-258	All X-258 except on fin III, #4 panel was D-100	All Normco 4551 on Fiberglass with loose outer coat of aluminized Fiberglass on Refrasil cloth.
SA-07	S-I-3	05/17/62	4:42 pm	31	30	Scheduled duration completed		Cutoff given after completion of gimbal program.	Eng. 1 2, 3, & 4	4-Coil	No new coating		
SA-08	S-I-3	05/24/62	4:41 pm	119	LOX depletion	LOX depletion		Flt. cutoff seq. initiated by LOX low level switch at 116.43 to inboard engine. Outboard engine cutoff came at 119.43.	Eng. 1 2, 3, & 4	4-Coil	No new coating	Same as test SA-06 & SA-07 except No. 4 panel on fin IV which was M-31 material	
SAT-25	SA-T4	06/19/62	4:46 pm	32	30	Scheduled duration completed	1600 Sec.	One heat exchanger (pos No. 3) was disconnected for GOX pressurization study.	Eng. 1, 2, 3, & 4	4-Coil	No insulation. No bubble covers	All ctd. with X-258except one panel which was D-100	All Normco 4551 on Fiberglass with loose outer coat of aluminized Fiberglass on Refrasil cloth excpt No. 3 was uncoated 3-layer Fiberglass
SAT-26	SA-T4	07/12/62	4:40 pm	12	120	Loss of lube oil press. measurement on eng. pos.3	1612 Sec.	SA-5 type GOX pressurizing valve utilized. 60 secs. LOX bubbling. No hypergol purge used.	None	3-Coils .105 orifice, 1 coil blank	No insulation. No bubble covers	All ctd. with X-258except one panel which was D-100	
SAT-27	SA-T4	07/13/62	4:40 pm	20	120	Loss of eng pos. 8 fuel pump inlet pressure measurement	1632 Sec.	SA-5 type GOX pressure izing valve utilized 60 secs. LOX bubbling. No hypergol purge used.	Eng. 3 & 4	3-coils .105 orifice, 1 coil blank	No insulation. No bubble covers	All ctd. with X-258 excpt one panel which was D-100	
SAT-28	SA-T4	07/17/62	4:40 pm	120	120	Duration	1752 Sec.	SA-5 type GOX pressurizing valve utilized 60 secs. LOX bubbling No hypergol purge used.	Eng. 3 & 4	3-coils .105 orifice, 1 coil blank	No insulation. No bubble covers	All ctd. with X-258 excpt which was D-100	

Table 7-1. Saturn I Static and Flight Test Information from 04/10/62 through 07/17/62 (Sheet 2 Of 2)

(TI 8 & HA)

TEST NO.	GG CONTROL VALVE	LOX SYSTEM PRESSURIZATION STUDY	LOX HIGH PRESSURE BLEED LINE CONFIGURATION	RATE GYRO LOCATION STUDY	TURBINE SPIN TYPE TEMP COND DUR	TURBINE SPINNER INITIATORS	GAS GENERATOR AUTO IGNITERS	HYPERGOL AND CONAX	MISCELLANEOUS
SA-06	Cracking check valves for all 8 engs.	GN2 pre-pressurization	½-inch lines routed below flowmeter	No	Type A-1 50°F for 72 hrs.	Fleming F61001-100, 500 volt	P/N 651139	HYP-650818 CON-1804001-01	
SA-07	Cracking check valves for all 8 engs.	GN2 pre-pressurization	½-inch lines routed below flowmeter	No	Type A-1 45°F for 72 hrs.	Fleming F61001-100, 500 volt	P/N 651139	HYP-650818 CON-1804001-01	
SA-08	Cracking check valves for all 8 engs.	Helium pre-pressurization	½-inch lines routed below flowmeter	No	Type A-1 45°F for 72 hrs.	Fleming F61001-100, 500 volt	P/N 651139	HYP-650818 CON-1804001-01	All LOX vent valves were held closed from X+95 to cutoff to evaluate the amount of pressurizing GOX supplied by the heat exchangers.
SAT-25	Yoke valves all 8 engs.	Helium pre-pressurization	½-inch lines rtd. below flowmeter except pos. No. 3 consisted of a proposed SA-5 line	No	Type A-1 45°F for 72 hrs.	Fleming F61001-100, 500 volt	P/N 651139	HYP-650818 CON-1804001-01	Position No. 4 heat shld curtai was of SA-5 configuration. Heat shld panel between chamber & heat exchanger coated with M 31
SAT-26	Yoke valves all 8 engs.	Helium pre-pressurization SA-5 ullage used	½-inch lines on all engines	No	Type A-1 45°F for 72 hrs.	Fleming F61001-100, 500 volt	P/N 651139	HYP-650818 CON-1804001-01	Position No. 4 heat shld curtai was of SA-5 configuration. Heat shld panel between chamber & heat exchanger coated with M 31
SAT-27	Yoke valves all 8 engs.	Helium pre-pressurization SA-5 ullage used	½-inch lines on all engines	No	Type A-1 45°F for 72 hrs.	Fleming F61001-100, 500 volt	P/N 651139	HYP-650818 CON-1804001-01	Position No. 4 heat shld curtai was of SA-5 configuration. Heat shld panel between chamber & heat exchanger coated with M 31
SAT-28	Yoke valves all 8 engs.	Helium pre-pressurization SA-5 ullage used	½-inch lines on all engines	No	Type A-1 45°F for 72 hrs.	Fleming F61001-100, 500 volt	P/N 651139	HYP-650818 CON-1804001-01	Position No. 4 heat shld curtai was of SA-5 configuration. Heat shld panel between chamber & heat exchanger coated with M 31

Table 7-1. Saturn I Static and Flight Test Information from 09/11/62 through 09/26/62 (Sheet 1 Of 2)

(T1.7B)

TEST NO.	VEHICLE NO.	DATE	TIME (OF DAY)	DURATION (SEC)	SCHEDULED DURATION (SEC)	REASON FOR CUTOFF	ACCUMUL. VHCL TIME SAT ONLY	REMARKS	GIMBAL	HEAT EXCHANGE	TAIL SHROUD	HEAT SHIELD	THROAT CURTAINS
SA-09	S-I-4	09/11/62	11:18 am	32	30	Scheduled duration completed		Cutoff given after completion of gimbal program.	Eng. 1 & 3	4-coil	No insulation. No bubble covers	All ctd with X-258	Pos. 1 & 2 Normco 4551 on Fiberglass with loose outer coating of aluminized Fiberglass or Refrasil cloth. Pos. 3 & 4 were same as Pos. 1 & 2 except Normco 4551 was replaced with a compound produced by Dow-Corning.
SA-10	S-I-4	09/26/62	4:42 pm	121	LOX depletion	LOX depletion		Flt. cutoff seq. initiated by LOX low level switch at 121.5. Eng. 1, 2, & 3 were cut off by the 6 second timer at 127.56 Eng. No. 4 cutoff given by thrust OK failure at 127.43 seconds.	Eng. 1, 3 & 4		No insulation. No bubble covers	All ctd with X-258	

Table 7-1. Saturn I Static and Flight Test Information from 09/11/62 through 09/26/62 (Sheet 2 Of 2)

(TI.88)

TEST NO.	GG CONTROL VALVE	LOX SYSTEM PRESSURIZATION STUDY	LOX HIGH PRESSURE BLEED LINE CONFIGURATION	RATE GYRO LOCATION STUDY	TURBINE SPIN TYPE COND TEMP DUR	TURBINE SPINNER INITIATORS	GAS GENERATOR AUTO IGNITERS	HYPERGOL AND CONAX	MISCELLANOUS
SA-09	Yoke valves all 8 engs.	GN2 pre-pressurization	[illegible]-inch lines routed below flowmeter	No	Type A-1 45°F for 72 hrs.	Fleming F61006-100, 500 volt	P/N 651139	HYP-650818 CON-1804001-01	
SA-10	Yoke valves all 8 engs.	Helium pre-pressurization	[illegible]-inch lines routed below flowmeter	No	Type A-1 45°F for 72 hrs.	Fleming F61006-100, 500 volt	P/N 651139	HYP-650818 CON-1804001-01	

Table 7-1. Saturn I Static and Flight Test Information from 10/26/62 through 03/27/63 (Sheet 1 Of 2)

(TI.9 & 9A)

TEST NO.	STAGE NO.	DATE	TIME (OF DAY)	DURATION (SEC)	SCHEDULED DURATION (SEC)	REASON FOR CUT OFF	ACCUMUL. STG TIME SAT ONLY	REMARKS	GIMBAL	HEAT EXCHANGER	TAIL SHROUD	HEAT SHIELD
SAT-29	SA-T4.5	10/26/62	4:49 pm	31	30	Duration	1783	LOX bubbling 30 secs.	None	One coil blocked off. Remaining 3 coils with 0.112" dia. orifice.	No insulation No bubble covers.	All X-258 except 8 panels which were M-31.
SAT-30	SA-T4.5	11/02/62	4:44 pm	65	65	Duration	1848	SA-5 tank ullage used. LOX bubbling time 60 seconds. GOX inter-connect orifice valves left open.	Eng. 1, 2,3, & 4	One coil blocked off. Remaining 3 coils with 0.112" dia. orifice.	No insulation No bubble covers.	All X-258 except 8 panels which were M-31.
SAT-31	SA-T4.5	11/09/62	4:42 pm	115	115	Duration	1963	SA-5 tank ullage used. LOX bubbling time 60 seconds. GOX inter-connect orifice valves left open.	Eng. 1, 2,3, & 4	One coil blocked off. Remaining 3 coils with 0.112" dia. orifice.	No insulation No bubble covers.	All X-258 except 8 panels which were M-31.
SA-11	S-1-5	02/27/63	4:47 pm	32	35 or Gimbal Program Completion	Scheduled Gimbal Program Complete		Cutoff given after completion of gimbal program. LOX bubbling 60 seconds. Engine H-5006 replaced Eng. H-5005 prior to test due to possible No.3 bearing problem.	Eng. 1, 2,3, & 4	Three coils 0.116" dia., 1 coil blank		All coated with M-31.
SA-12	S-1-5	03/13/63	4:17 pm	141	LOX depletion	LOX depletion		Flt. cutoff initated by LOX low level switch at 135.44 secs. to inboard engs. Outboard eng. cutoff came at 141.44 sec. LOX bubbl-ing 60 seconds.	Eng. 1, 2,3, & 4	Three coils 0.116" dia., 1 coil blank		All coated with M-31.
SA-13	S-1-5	03/27/63	4:38 pm	143	LOX depletion	LOX depletion		Flt. cutoff initiated by LOX low level switch at 137.47 seconds to Inboard engs. Outboard engs. cutoff came at 142.47 sec. LOX bubbl-ing 60 seconds.	Eng. 1, 2,3, & 4	Three coils 0.108" dia. 1 coil blank		All coated with M-31.

Table 7-1. Saturn I Static and Flight Test Information from 10/26/62 through 03/27/63 (Sheet 2 Of 2)

(TI.10 & 10A)

TEST NO.	THROAT CURTAINS	LOX SYSTEM PRESSURIZATION STUDY	LOX HIGH PRESSURE BLEED LINE CONFIGURATION	TURBINE SPINNER TYPE-CONDITIONED TEMP-DURATION	TURBINE SPINNER INITIATORS	GAS GENERATOR AUTO IGNITERS	HYPERGOL AND CONAX	MISCELLANOUS
SAT-29	Al Normco 4551 on fiberglass with loose outer coat of alumnized fiberglass or refrasil cloth except No. 3 which was uncoated, 3-layer fiberglass. Eng. pos. No. 2 curtain was SA-5 configuration.	GN2 pre-pressurization	¾" lines on all engines.	Type A-2 45°F for 72 hours	Fleming F61001-100, 500 volt	P/N 651139	HYP-651150 CON-1804001-01	First test with 188K engs. SA-5 inboard turbine exhaust duct configuration on eng. position No. 8. 4-inch GOX line failed prior to test. New GOX line installed. All engines with SA-5 type upper turbine exhaust duct.
SAT-30		Helium pre-pressurization	¾" lines on all engines.	Type A-2 45°F for 72 hours	Fleming F61001-100, 500 volt	P/N 651139	HYP-651150 CON-1804001-01	SA-5 inboard turbine exhaust duct configuration on eng. pos. No. 8 All engines with SA-5 type upper turbine exhaust duct.
SAT-31		Helium pre-pressurization	¾" lines on all engines.	Type A-2 45°F for 72 hours	Fleming F61001-100, 500 volt	P/N 651139	HYP-651150 CON.-1804001-01	
SA-11	All Normco 4551 on fiberglass with loose outer coat of aluminized fiberglass on refrasil cloth.	Helium pre-pressurization	¾" lines on all engines.	Type A-2 45°F for 72 hours	McCormick-Selph 804439, 500 volt	P/N 65139	HYP-650818 CON-1804001-01	Fuel leakage occurred around the turbopump Calrod heating unit on engine positions No. 2, 4, 5, 6, and 8. 7-inch LOX vent position micro-switch malfunctioned during countdown.
SA-12	All Normco 4551 on fiberglass with loose out coat of aluminized fiberglass on refrasil cloth.	Helium pre-pressurization	¾" lines on all engines.	Type A-2 45°F for 72 hours	McCormick-Selph 804439, 500 volt	P/N 65139	HYP-650818 CON-1804001-01	LOX pressurization system exhibited unstable conditions during the test. 7-inch LOX vent position micro-switch malfunctioned during countdown. Turbine to gear-case seal leaks discovered post-test. LOX pump seal leak eng. pos. No. 5 Eng. pos. No. 4 thrust high.
SA-13	All Normco 4551 on fiberglass with loose out coat of aluminized fiberglass on refrasil cloth.	Helium pre-pressurization	¾" lines on all engines.	Type A-2 45°F for 72 hours	Fleming F61006-100, 500 volt	P/N 65139	HYP-650818 CON-1804001-01	Heat exchangers re-orificed and adjustments were made to the GOX flow control vlv prior to test. All turbine to gearcase seals were changed prior to test; however, post-test leak checks revealed leaks at all engs. 7-inch LOX vent position microswitch malfunctioned during countdown.

Table 7-1. Saturn I Static and Flight Test Information from 10/02/63 through 10/22/63 (Sheet 1 Of 2)

(TI.11)

TEST NO.	STAGE NO.	DATE	TIME (OF DAY)	DURATION (SEC)	SCHEDULED DURATION (SEC)	REASON FOR CUT OFF	REMARKS	GIMBAL	HEAT EXCHANGER	HEAT SHIELD
SA-16	S-I-7	10/02/63	4:38 pm	33.8	35	Duration	Scheduled cutoff to inboard engs. initiated at 33.78 seconds by the firing panel operator. Outboard engine cutoff came at 33.88 secs. LOX bubbling 50 secs. Engine H-5014 (Pos. No. 2) experienced considerable fuel leakage from thrust chamber tubes above the throat in the combustion zone.	Engs. 1 2, 3, and 4	One coil blank Three coils 0.107 in. dia.; Except eng. positions 6 & 8. Position 6; two coils 0.108 inch dia., one coil 0.107 inch dia. Position 8; three coils 0.108 inch dia.	All coated with M-31
SA-17	S-I-7	10/22/63	4:38 pm	145.22	LOX depletion	LOX depletion	Flt. cutoff initiated by LOX low level sensor at 140.21 secs. to inboard engines. Outboard eng. cutoff came at 145.22 secs., initiated by the thrust ok pressure switch from eng. pos. No. 3 LOX bubbling 50 secs. Engine H-5014 (Pos. No. 2) replaced by H-5027 prior to this test.	Engs. 1 2, 3 and 4	One coil blank Three coils 0.107 in. dia.; Except eng. positions 6 & 8. Position 6; two coils 0.108 inch dia., one coil 0.107 inch dia. Position 8; three coils 0.108 inch dia.	All coated with M-31

Table 7-1. Saturn I Static and Flight Test Information from 10/02/63 through 10/22/63 (Sheet 2 Of 2)

(TI.12)

TEST NO.	THROAT CURTAINS	TURBINE SPINNER TYPE & CONDITIONED TEMP & DURATION	TURBINE SPINNER INITIATORS	GAS GENERATOR AUTO IGNITERS	HYPERGOL AND CONAX	MISCELLANOUS
SA-16	All Normco 4551 on fiberglass with loose outer coat of aluminized fiberglass on refrasil cloth.	Type A-2, 45°F for 72 hours.	Fleming F61006-100 500 volt.	P/N 651139	HYP. - 651150 CON. - 1804001-01	Engine H-5014 was replaced after this test by H-5027 due to cracked tubes inside the chamber in the combustion zone. Excessive fuel leakage resulted from the cracks in the tubes.
SA-17	All Normco 4551 on fiberglass with loose outer coat of aluminized fiberglass on refrasil cloth.	Type A-2, 45°F for 72 hours.	Fleming F61006-100 500 volt.	P/N 651139	HYP. - 651150 CON. - 1804001-01	A leak from tube to outside was discovered at Pos. No. 8 (Engine H-2015), on the thrust chamber above the throat. The leakage passed through a crack in the weld on the outside of the chamber. The engine will be replaced after removal from the test stand.

Table 7.2. Saturn SI/IB Static Test Information

STAGE NUMBER	INSTALLED	REMOVED	TIME AT SITE	TEST NO.	DATE OF TEST	DURATION (SEC.)	SCHEDULED DURATION (SEC.)	REMARKS
S-1-1	March 6, 1961	May 25, 1961	11.4 Weeks	SA-01 SA-02 SA-03	04/29/61 05/05/61 05/11/61	30 44 111	30 LOX depletion LOX depletion	Stage delivered to STT lacking complete assembly. LOX system flushed, due to contamination on the LOX pump inlet screens. Test SA-02 terminated prematurely by hot gas leak.
S-1-2	Sept. 19, 1961	Nov. 2, 1961	6.3 Weeks	SA-04 SA-05	10/10/61 10/24/61	32 119	30 LOX depletion	
S-1-3	March 19, 1962	May 31, 1962	10.4 Weeks	SA-06 SA-07 SA-08	04/10/62 05/17/62 05/24/62	30 30 119	30 30 LOX depletion	During test SA-06 the gearcase composite vibration level of engine position No. 2 was abnormally high. Disassembly revealed a damaged LOX pump bearing. Unusual spectrum analysis (frequency vs "g" r.m.s. level) of the gearcase vibration data from engine positions No. 4, 5, 6, and 8 prompted subsequent engine removal and turbopump inspection. Turbopump disassemblies necessitated a second short duration test.
S-1-4	Aug. 1, 1962	Oct. 1, 1962	8.7 Weeks	SA-09 SA-10	09/11/62 09/26/62	32 121	30 LOX depletion	LOX system flushed due to contamination on the LOX pump inlet screens. Contamination discovered throughout the facility GN2 supply system and on the stage. Systems cleaned and test SA-09 performed. Contamination again discovered, and stage and facilities were again cleaned.
S-1-5	Jan. 28, 1963	Apr. 2, 1963	9.1 Weeks	SA-11 SA-12 SA-13	02/27/63 03/13/63 03/27/63	32 141 143	35 LOX depletion LOX depletion	Engine position No. 4, H-5006, replaced with H-5005, prior to test SA-11 due to suspected No. 3 bearing problems. LOX pressurization system exhibited unstable conditions during test SA-12. Heat exchangers re-orificed and adjustments made to GOX flow control valve prior to test SA-13.
S-1-6	Apr. 22, 1963	June 17, 1963	8 Weeks	SA-14 SA-15	05/05/63 06/06/63	34 136	35 LOX depletion	Engine position No. 8 (H-2009), main LOX valve failed to fully close during the cutoff phase of test SA-14. This resulted in extensive engine damage, requiring replacement of this engine with engine No. H-2007.
S-1-7	Sept. 6, 1963	Nov. 4, 1963	8.4 Weeks	SA-16 SA-17	10/02/63 10/22/63	34 139	35 LOX depletion	Engine position No. 2 (H-5014) was replaced with engine H-5027 after test SA-16 because of excessive thrust chamber tube leakage. External fuel lube leakage was noted on engine position No. 8 (H-2015) after test SA-17. This engine was replaced with engine (H-2021) after removal of the stage from the test stand.
S-1-9	Feb. 17, 1964	Apr. 8, 1964	7.1 Weeks	SA-18 SA-19	03/13/64 03/24/64	35 142	35 LOX depletion	An auxiliary LOX dome purge manifold was installed in order to prevent inboard engine dome contamination during outboard cutoff. This purge did not completely eliminate the contamination problem.
S-1-8	Apr. 27, 1964	June 23, 1964	8.1 Weeks	SA-20 SA-21	05/26/64 06/11/64	49 140	45 LOX depletion	Six engines were re-orificed after test SA-20 due to high thrust level. During test SA-21, engine position No. 8 (H-2029) had two downward performance shifts. These shifts were evident in each parameter except conisphere temperature. Engine position No. 8 (H-2029) was replaced with engine (H-2031). Upon disassembly of the turbine from engine (H-2029), the following discrepancies were noted which can be contributed to the performane shifts. a. The 1-B turbine seal retainer was broken in four places. b. The blades on the first stage turbine were worn and defaced. Position No. 6 (H-2017) was removed from the cluster to investigate the thrust bias between Neosho tests and stage tests. This engine was replaced with engine (H-2032) after the stage returned to Michoud.

These objectives were evaluated and specific objectives to be satisfied during each individual tests were carefully factored into the test plan for that test. A total of 31 development static firings were made during the S-I/S-IB program.

S-I/S-IB Test Accomplishments

The major accomplishment contributed by the Saturn SI/S-IB program all-system firings was verification that the clustered engine and tank concept for providing large thrust launch vehicles was feasible.

Specific accomplishments obtained from the S-I/S-IB propulsion system test program are presented in the following paragraphs. Each issue has been classified as to consequence and time phasing for which the issue may apply as was done in Appendix 3 for Space Shuttle.

Engine Purge Sequence (workable, mod. expected, flight). The H-I engine used a hypergolic substance to lead the RP-1 fuel into the combustion chamber to establish ignition with the liquid oxygen before the fuel entered the chamber. A cutoff sequence had been established which initiated a hypergol purge at 100 milliseconds after cutoff signal. It was determined that the 100 millisecond purge was forcing hypergol into the main fuel valve actuators on the four inboard engines resulting in slow closing times for the valves. The hypergol purge sequence was changed to two seconds after cutoff which eliminated the slow main fuel valve closing problem.

Ignition Delays on Solid Propellant Gas Generators (unworkable, preflight). The H-I engine used a solid propellant gas generator (SPGG) for the initial spin-up of the turbopumps prior to liquid oxygen/RP-1 ignition in the gas generators. During the test program, problems with SPGG ignition delays, or hang fires, was experienced. It was determined that the delays were caused by igniter pellet shifting due to a weak cover. The cover was redesigned and the problem solved.

Fuel Temperature Stratification (catastrophic, preflight). The H-I engine start sequence was extremely sensitive to mixture ratio of propellants during gas generator ignition. A mixture ratio that was too liquid oxygen rich could be destructive to the turbopump turbine blades. Controlling the temperature of the fuel at the pump inlet was vital. It was determined that temperature stratification of the fuel could be prevented by bubbling the fuel with gaseous nitrogen injected into the fuel feedline for each engine.

Liquid Oxygen Pump Inlet Temperatures (catastrophic, preflight). The Liquid oxygen pumps on the H-I engine were extremely sensitive to NPSH and the gas generator was sensitive to ignition mixture ratio. It was necessary to control the liquid oxygen pump inlet temperature within a narrow band to satisfy both requirements. It was determined that the liquid oxygen temperature could be kept within the required start box by bubbling the liquid oxygen feedlines with helium for ten seconds prior to ignition signal.

Engine Compartment Heating (catastrophic, preflight). The S-I/S-IB engine compartment contained components which were sensitive to the temperature range at which they operated. The primary temperature sensitive components were the SPGGs and the MFV actuators. It was necessary to control the temperature in the engine compartment to assure proper functioning of these components. An engine compartment heating system was developed and demonstrated to have the capability to maintain satisfactory temperatures for all critical components.

Turbine Bearing Seizure (unworkable, flight). The H-I engine turbopump gear box used RP-1 fuel for a bearing lubricant. To improve the lubricity of the RP-1, oronite was added to the fuel. During the test program, difficulty was experienced with turbine bearing seizure during post test turbopump torque checks. It was determined that this problem was caused by coking of the oronite at the temperatures of the numbers 7 and 9 bearings. The oronite additive was eliminated from the numbers 7 and 9 bearing lubricant which solved the problem.

Engine Compartment Heat Shield (workable, mod. expected, flight). A requirement existed to protect the components mounted in the engine compartment from the heat generated by the eight clustered H-I engines. During the test program, experimentation with various coatings established the optimum coating material to be used on the heat shield.

Liquid Oxygen Vent and Relief Valve Position Indicators (unworkable, preflight). A problem with the stage liquid oxygen vent and relief valve position indicator micro-switches failing to indicate the valve position properly was found to be caused by the cryogenic conditions the valve was being subjected to during static tests. The valve had been qualified at ambient conditions. The valve was re-qualified under cryogenic conditions and no more problems were experienced.

Liquid Oxygen Tank Ullage Collapse (unworkable, flight). During static firing, the stage liquid oxygen tanks were prepressurized with either gaseous nitrogen or helium. On the early tests, ullage pressure collapse was experienced during prepressurization. The initial design of the liquid oxygen pressurizing system did not include a gas diffuser at the inlet to the tank. A redesign was made which incorporated a diffuser and subsequent performance of the pressurizing system was satisfactory.

Damage to Liquid Oxygen Tank Standpipes (unworkable, flight). The S-I/S-IB stage configuration used four 70-inch diameter liquid oxygen tanks clustered in a circle around a central 105-inch diameter liquid oxygen tank. During propulsion system operation, liquid oxygen was transferred from the central tank to the outboard tanks through a standpipe located in each tank. The original design of the standpipes had them open on the discharge end. A sudden liquid oxygen tank ullage pressure decrease occurred mid-way into a firing. Pump NPSP requirements were violated and aluminum standpipes failed structurally. The cause of the ullage pressure change was spraying liquid oxygen from the standpipes into ullage gas and subsequent pressurant condensation. A redesign incorporated covers for the standpipes and resolved the issue.

Damage to Engine Compartment Heat Shield Flexible Curtains (unworkable, flight). The design of the engine compartment heat shield included a flexible curtain fitted around the four outboard H-I engines which were designed to gimbal for thrust vector control. Several failures occurred during static tests, indicating a weakness in the design of the flexible curtains. A redesign of the curtains followed by extensive development testing precluded further failures.

Thrust Structure Weakness (catastrophic, flight). The first TVC gimbal test performed on the S-I static test vehicle uncovered matching frequencies between the stage thrust structure and the gimballed engines. This forced a redesign of the thrust structure to decouple the frequencies.

H-I Engine Thrust Increase (improvement, flight). During the course of the S-I/S-IB program, the H-I engine thrust was increased in increments from 165,000 pounds to 205,000 pounds. Each thrust level was satisfactorily demonstrated to be feasible for H-I engine operation in the clustered configuration by the propulsion system test program.

Liquid Oxygen Pump Seal Failure (catastrophic, flight). During test SA-52, a liquid oxygen pump seal failure caused fire and explosion in engine position 8. The following is quoted from the failure analysis report:

> "Had this been a launch and the same progression of events occurred, the launch fire detection system would not have prevented launch. Such an instance would have most probably resulted in loss of this engine in flight. Loss of the stage and mission is debatable since prevalves are provided which could control damage provided the loss of this engine was mild enough so that damage was confined to only that engine.
>
> "It is also noteworthy that had this failure occurred in flight and progressed to the point of engine or stage loss, it would not have been possible with the present flight instrumentation to have isolated the cause of the gross failure, much less the specific source of failure."

Following this failure, the S-IB liquid oxygen seal was replaced with a bellows type seal. No additional malfunctions were noted.

Turbine Blade Failure (catastrophic, flight). A potentially catastrophic turbine blade failure occurred during Test SA-40. Subsequent inspection determined that five H-1 engines had turbine blades manufactured from incorrect material. Turbine wheels were replaced on all of these engines, and they were re-tested at the manufacturer's site.

APPENDIX 8

SATURN I, S-IV MAIN PROPULSION SYSTEM

S-IV (Figure 8-1) was the second stage of the Saturn I vehicle and was developed by Douglas Aircraft Company for NASA. Development began in April 1960 with the initial flight on January 29, 1964, and the final flight, number 6, on July 30, 1965. Payload to orbit capability was approximately 34,000 pounds. All six flights were successful. This stage served as the basis for the larger, more powerful S-IVB stage of the Apollo program. Figures 8-2 through 8-5 provide insight relative to stage design and important stage systems.

The stage was propelled by six Pratt and Whitney-developed RL 10-A-3 engines which used liquid hydrogen and liquid oxygen as propellants. The six engines were configured in a circle and thrust vector control was accomplished by engine gimballing. Total vehicle thrust was 90,000 pounds, approximately 15,000 pounds per engine. The engines were ignited after S-I booster stage burn completion, approximately two minutes after liftoff, and then burned for approximately eight minutes. The engines operated at a nominal mixture ratio of 5:1 and were ignited only once, although they were capable of multi-starts.

The vehicle diameter was 22 feet. Liquid hydrogen propellant was located forward in a cylindrical single tank—unlike the booster stage which used multi tanks for each propellant—while liquid oxygen was located aft in an elliptical shaped single tank. The two propellants were separated by a single bulkhead, commonly known as common bulkhead. The bulkhead was a complex structure that not only physically separated propellants, but also provided thermal insulation and a means for detecting the presence of either of the propellants within the bulkhead interior—a serious safety concern if detected. Heat inputs to the liquid hydrogen tank sidewalls and forward bulkhead were controlled by internal foam insulation with an impervious barrier separating liquid hydrogen and the insulation. Helium gas stored at high pressure in spheres located within the liquid hydrogen tank was available for liquid oxygen tank pressurization and other purposes.

Propulsion system testing included both battleship and all system vehicles. Other test articles were available for structural, dynamics, and other test requirements. A total of 27 firings and total firing time of 5440 seconds were recorded for the Battleship configuration during the time interval of December 11, 1961 to January 26, 1963. No tests were accomplished on the all-systems configuration before its destruction on January 24, 1964.

RL 10A-3 Rocket Engine (Figures 8-2 and 8-3). Electrical signals controlled pressurized helium which was used to actuate valves for starting and stopping the engines. Ignition of the propellants was accomplished by an electrical ignition system. Each engine was equipped with a thrust control assembly that was a servo-operated, variable position valve which controlled engine thrust by regulating the amount of fuel bypassing the turbine, thereby controlling turbine speed and fuel pumping. Thrust chamber tubular construction permitted use of a

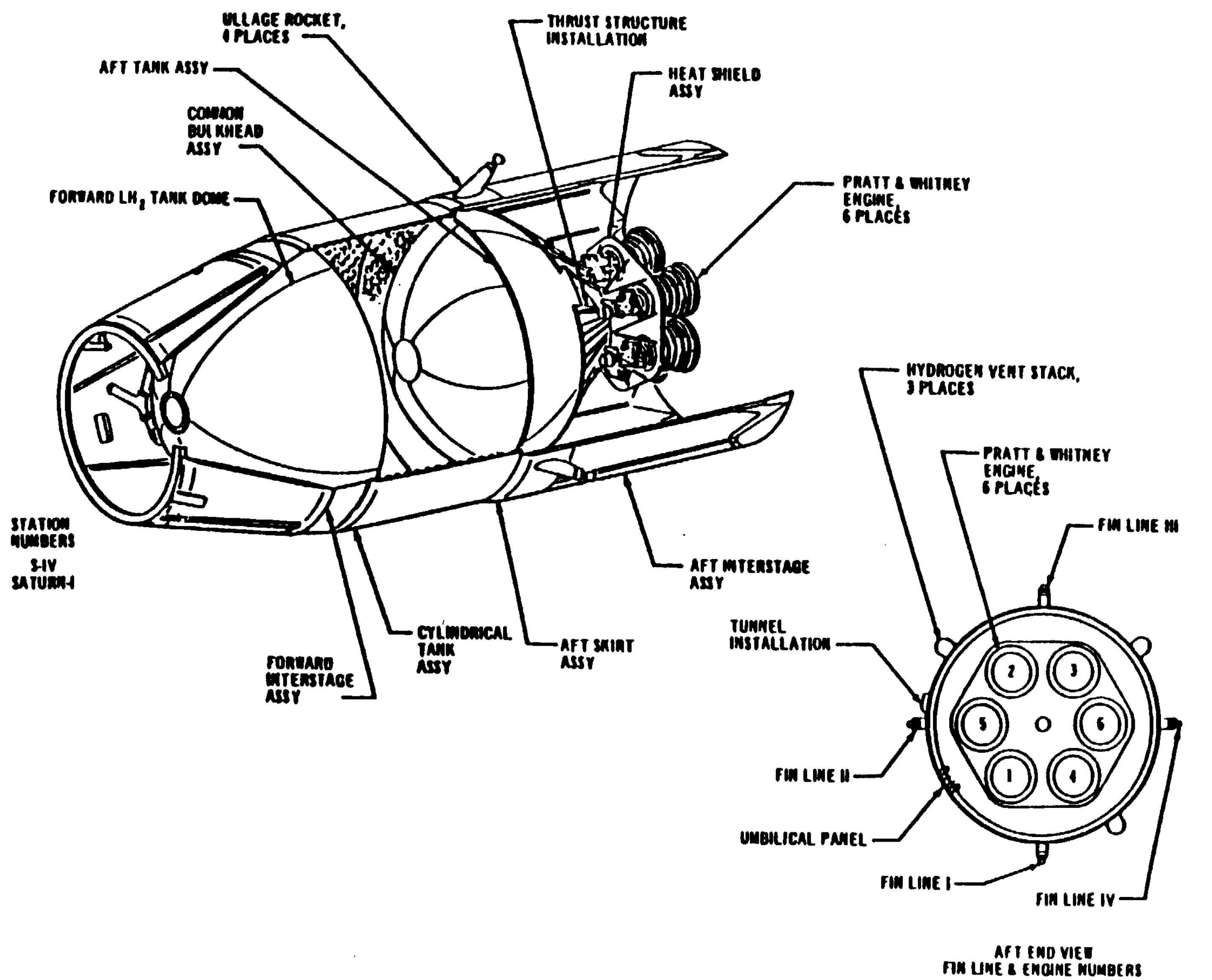


FIGURE 8-1. SATURN S-IV STAGE

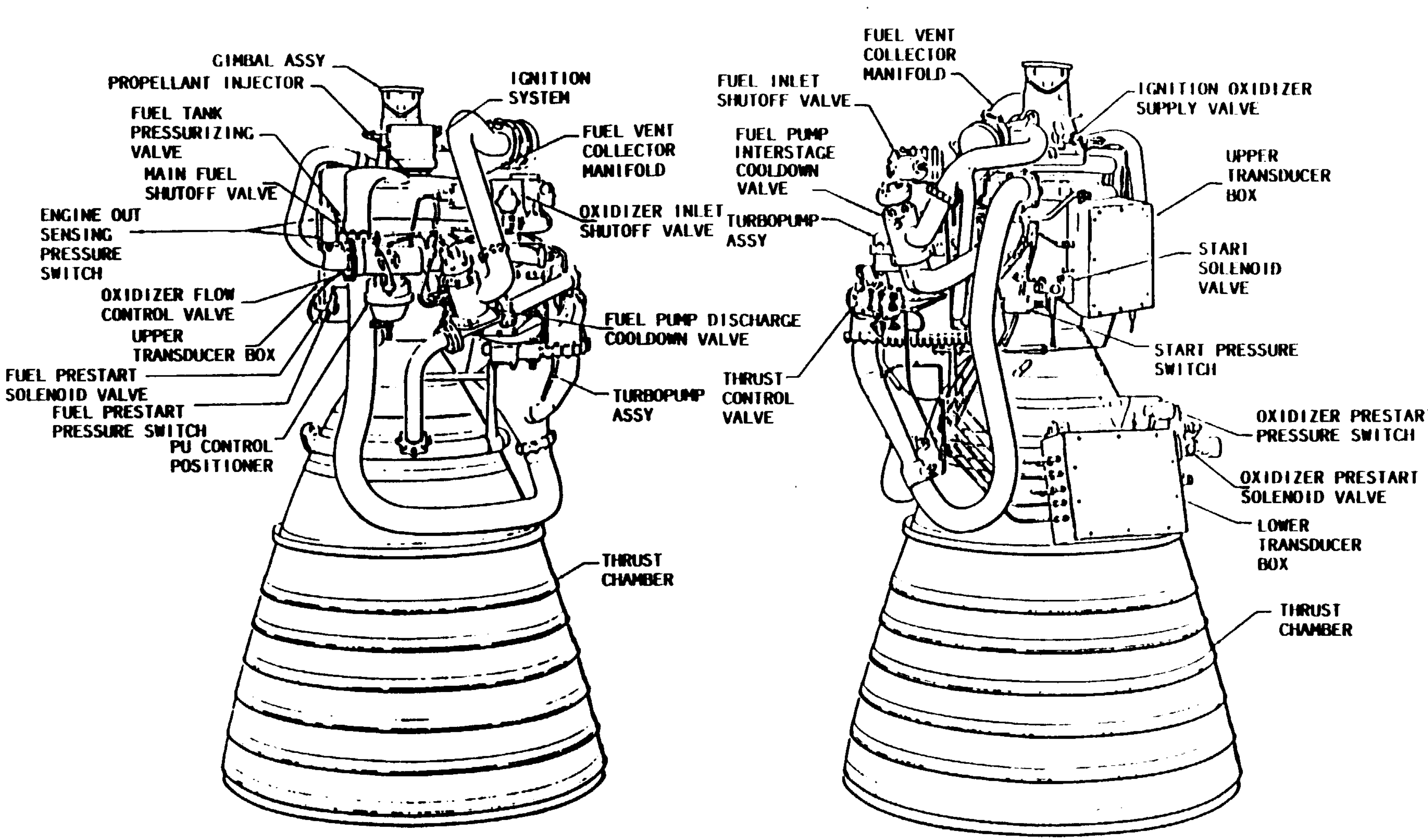


FIGURE 8-2. RL10A-3 COMPONENT LOCATIONS

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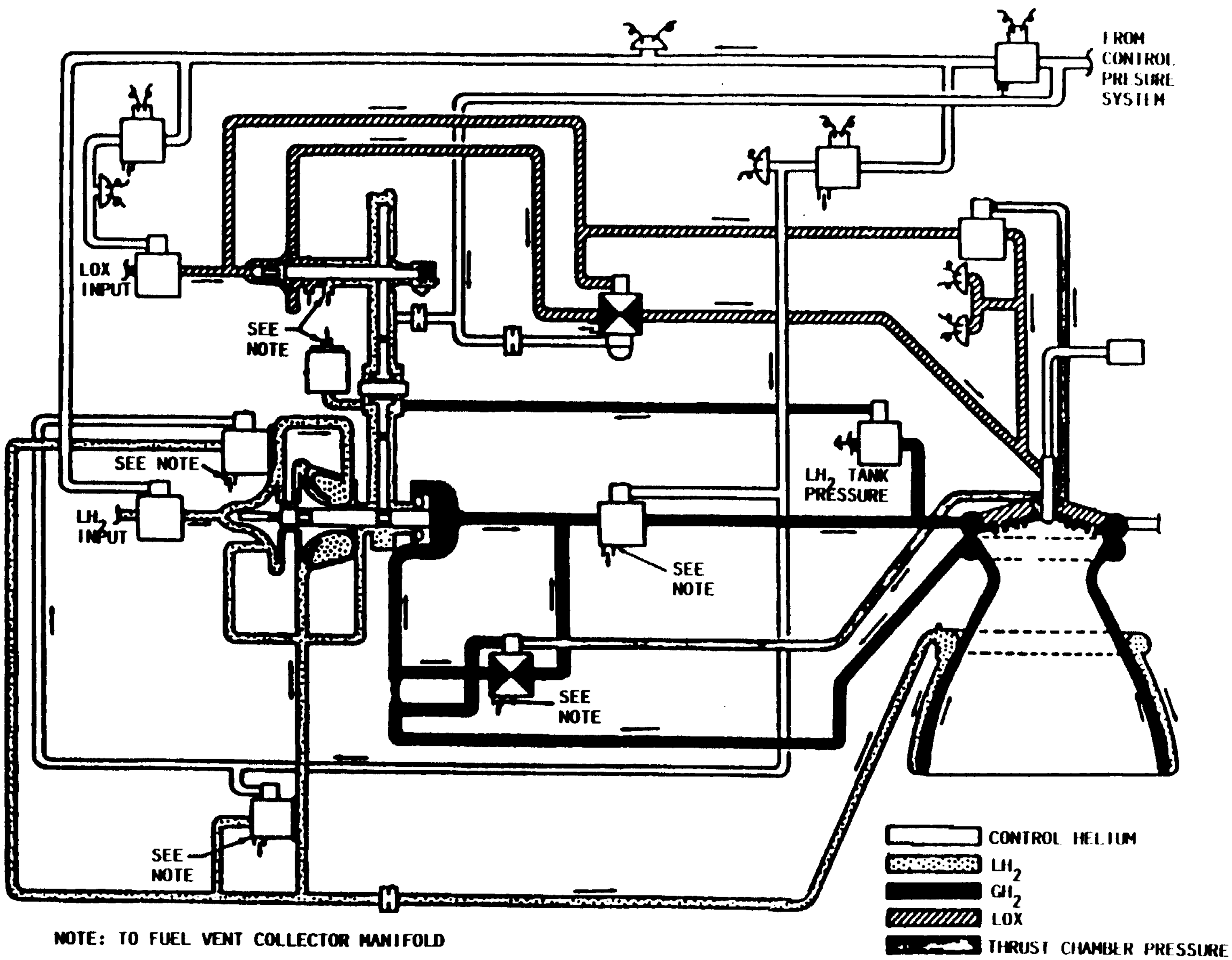

FIGURE 8-3. RL10A-3 ENGINE FUNCTIONAL SCHEMATIC

regenerative operating cycle. Liquid hydrogen used to cool the engine during operation, was turned into a gaseous state and re-used to drive the fuel and oxidizer turbopump and supply pressurant to the liquid hydrogen tank. More detail on the engine follows.

Major Components	Thrust Chamber, Fuel and Oxidizer Turbopump Assembly, Oxygen Flow Control Valve, spark Ignition Subystem, Thrust Control Assembly, Miscellaneous Control Valves.
Thrust Chamber	Tubular-walled with chamber stiffeners; expansion area ratio, 40 to 1; chamber pressure, 300 psia; liquid oxygen to liquid hydrogen mass burn ratio, 5:1.
Turbopump Assembly	Located outside thrust chamber at nozzle throat; turbine driven fuel and oxidizer pumps; turbine power supplied from expanding fuel.
Thrust Control Assembly	Mounted on turbopump assembly; controlled thrust chamber pressure by regulating turbine speed; servo supply pressure, 672 psi.
Oxidizer Flow Control Valve	Mounted on turbopump near nozzle throat of thrust chamber; controlled oxidizer-to-fuel mass ratio for proper ignition, and consumption of oxidizer to minimize propellant residual at burnout; mechanical stops limited variance.
Spark Ignition Subsystem	Consisted of oxidizer supply valve and ignition system; system used hydrogen from cool down tubes for fuel; oxidizer valve regulated oxygen supply to combustion chamber.

Stage Oxidizer System (Figure 8-4). Liquid oxygen was supplied to each engine by individual feedlines connecting each engine to a common sump located on the liquid oxygen tank bottom. A screen/vane assembly located at the tank outlet prevented liquid oxygen vortexing and provided filtration. Prior to launch, cold helium gas was injected into feedlines to cool the oxygen sufficiently to satisfy engine start requirements. Engine hardware cooling was accomplished during first stage powered flight by a regulated flow of liquid oxygen through the engine and exiting the nozzle. Part of this dumped flow was liquid which could solidify during the expansion to low pressures experienced at altitude. This was an unsafe event, thus a specially designed system using nitrogen prevented oxygen from solidifying.

Gaseous helium pressurized the liquid oxygen tank ullage immediately prior to launch. Cold helium gas from storage spheres within the hydrogen tank was heated in a small hydrogen/oxygen burner, called a helium heater, and used to pressurize the oxygen tank ullage immediately prior to and during S-IV powered flight phases. Tank located pressure-sensing switches provide the control for all phases of oxygen tank pressurization—both preflight and flight.

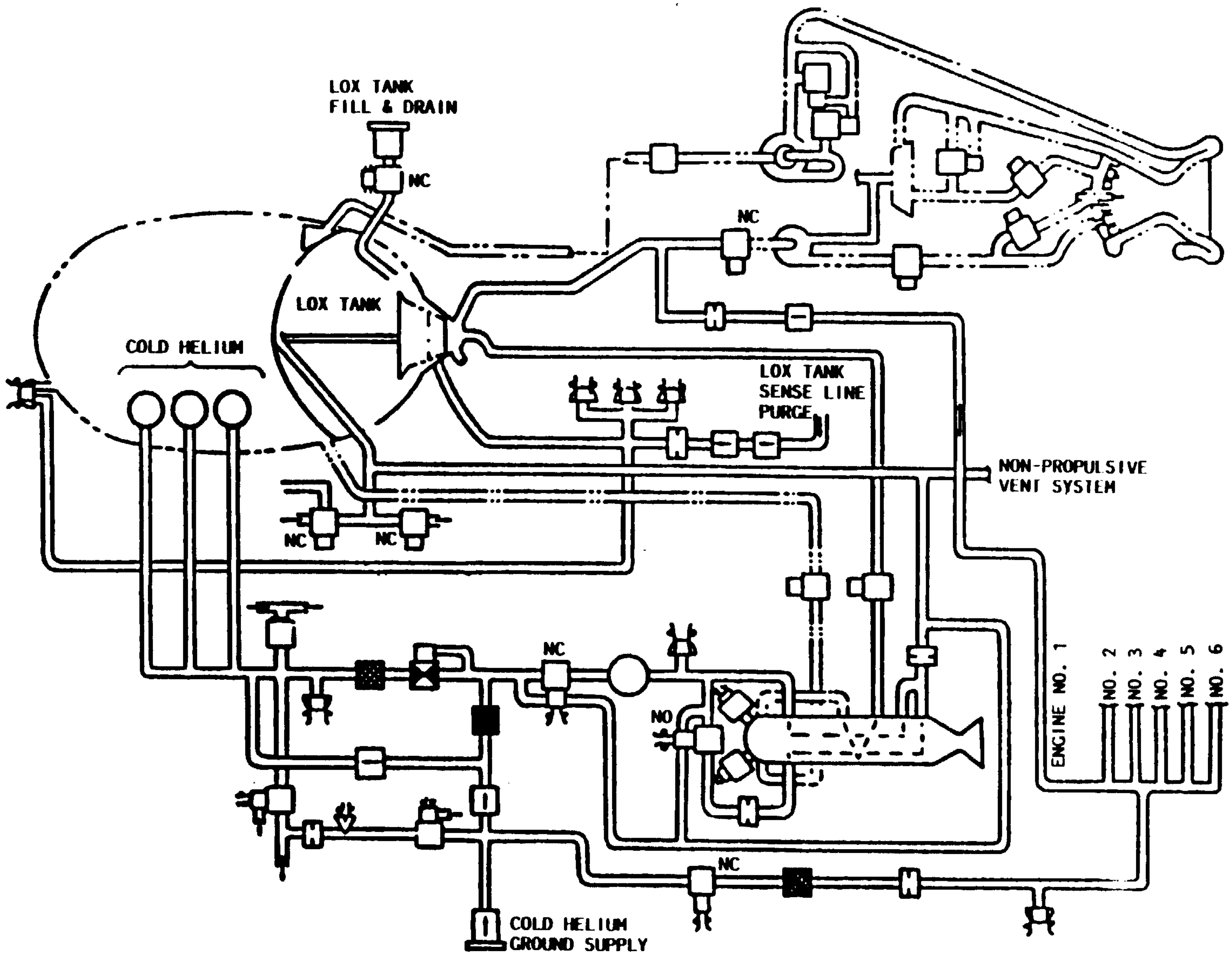

FIGURE 8-4. S-IV OXIDIZER SYSTEM FUNCTIONAL SCHEMATIC

A vent system provided for relief of pressure during fill and mainstage operation. The oxygen tank vent system was a combination vent and relief system. Two valves vented overboard all products of vaporization resulting from ground filling, and automatically relieved excessive pressure buildup within the tank during other operational phases. The vent valves operated independent of the tank pressure-sensing switches.

Stage Fuel System (Figure 8-5). Liquid hydrogen was supplied to each engine by individual feedlines from the liquid hydrogen tank. Each of the feedlines, total of six, was located external to the vehicle along the length of the liquid oxygen tank and penetrated the aft interstage assembly before connecting with the respective engine. Each feedline had appropriate flex and gimbal joints for maintaining structural integrity. Each feedline used vacuum jacket hardware for controlling heat input and anti-vortex devices and screens at the feedline tank interface. Hydrogen flow, necessary to cool engine hardware during first stage powered flight, was collected from each engine and dumped into three large lines attached to the outer vehicle skin. The lines ran the length of the vehicle and terminated outside the vehicle boundary layer in free stream flow near the vehicle base. This was necessary to prevent hydrogen gas entrainment and burning adjacent to the vehicle exterior.

Gaseous helium was used to pressurize the liquid hydrogen tank ullage during terminal count immediately prior to launch and immediately prior to stage engine start, approximately three minutes later. Hydrogen gas bled from the six engines pressurized the tank ullage during the engine operational phase. Tank located pressure-sensing switches provided the control function for all phases of liquid hydrogen tank pressurization—both preflight and flight.

A vent system provided for relief of ullage pressure during fill and mainstage operation and had similar operational characteristics as described for the oxygen tank.

Note: Reference documents on S-IV stage were unavailable, thus the above description and section on accomplishments presented later are reconstructed from memory.

S-IV ACCOMPLISHMENTS

Accomplishments are from the memory of personnel involved in the program and are thus incomplete. Reference documentation was not available.

Hydrogen Tank Insulation Cracking (unworkable, preflight). The liquid hydrogen tank of S-IV stage used internal insulation for controlling heat input to the propellant and to control frost/ice formation on the tank wall external surface. The insulating material for S-IV battleship was balsa wood. The all systems vehicle used the planned flight vehicle insulation—light weight foam blocks bonded to the internal wall and tank upper bulkhead. A light weight fiber glass cloth bonded to the insulation inner surface and coated with a sealant, epoxy was used to prevent insulation contamination with hydrogen. The initial propellant loading test of the all

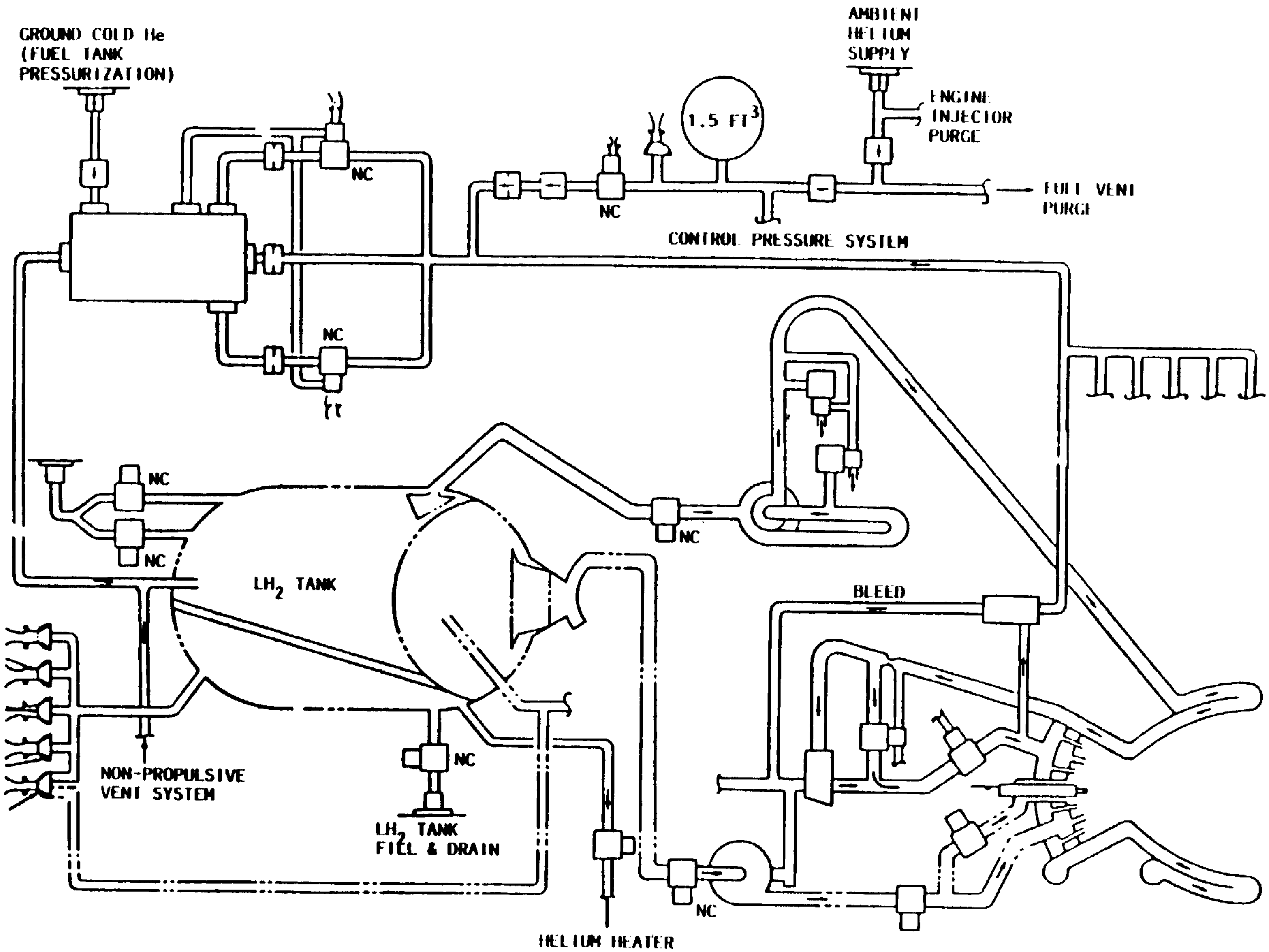

FIGURE 8-5. S-IV FUEL SYSTEM FUNCTIONAL
SCHEMATIC

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systems vehicle resulted in significant tank deflections and extensive insulation cracking. Testing was halted and extensive field repairs were made.

Hydrogen/Oxygen Common Bulkhead Leaking (catastrophic, preflight/flight). The all systems vehicle on the next test—another propellant loading test six weeks later—experienced an hydrogen leak in the common bulkhead—a potentially serious safety problem because of the close proximity of the oxygen tank. The vehicle was removed from the test stand for inspection and repair.

Engine Oxygen and Hydrogen Propellant Chill Disposal (unworkable, flight). The RL10 hydrogen/oxygen engine required hardware cool-down prior to engine start. The requirement was satisfied by controlled dumping of propellants through the engine during significant parts of first stage operation. Hydrogen gas disposal was through three large ducts attached to the vehicle skin which ran the length of the vehicle and then outward—dumping hydrogen in a free stream flow outside the vehicle boundary layer. Oxygen was dumped through the engine nozzle into the S-I/S-IV inter stage. Exiting liquid oxygen, present during some phases of the dump, expanded to near inter stage pressure and solidified. This was deemed to be hazardous, thus a system to prevent oxygen solidification was necessary and was developed. The system using large quantities of nitrogen within the inter tank was developed as a subsystem, verified in vacuum operation and integrated/verified in propulsion system testing.

All System Vehicle Destruction (catastrophic, preflight/flight). The oxygen tank of the all system vehicle was overpressurized during the first attempted static firing. An explosion resulted and the vehicle was a total loss. The test stand and ground support equipment were badly damaged.

Helium Heater Development Issues (unworkable, flight). A combustion device using gaseous oxygen and hydrogen propellants named helium heater was developed and used to heat cold He gas for flight pressurization of the oxygen tank. Serious subsystem problems were encountered in development of the helium heater. Hardware was unavailable for propulsion system testing when initially required and when used was unreliable. System data was needed to verify the total quantity of helium required, the required operational He flow rate range for the helium heater, establish compatibility of operation of tank pressurization control module and helium heater, etc.

Propulsion Hardware Failures (unworkable, preflight/flight). Vacuum jacketed feedlines, feedline expansion and gimbal devices, pressurization system flow control modules and tank pressure sensing devices, tank ullage pressure vent and relief valves all experienced frequent failures during propulsion system testing and required servicing and/or changeout. Failures resulted in test cancellation and delayed testing. Many of the problems experienced resulted from incomplete hardware development as components at the time required for system testing.

APPENDIX 9

TITAN PROGRAM

The Titan program had its inception in the middle 1950's with the development of the Titan I ICBM which, while operational, had the greatest range and payload in the U. S. Missile Arsenal. Titan I was almost immediately redesigned to the Titan II configuration which differed from its predecessor in two very advantageous respects: (1) it used storable propellants; and (2) it was launched directly from the silo. Titan II also had increased thrust and propellant weight giving it increased range and payload over Titan I. The booster for the Titan II ICBM served, in an essentially unchanged configuration, as the launch vehicle for Gemini spacecraft and as the liquid propellant core (Stages I and II) for Titan III. In this way, all the extensive vehicle testing and manufacturing experience was directly applicable to space operations. Testing activities were limited to only those things that changed. Minor changes for improvement in manufacturing or operations were easily evaluated. "Certified by similarity" became an accepted technique. Figures 9-1 and 9-2 illustrate the propulsion system for stages 1 and 2, respectively, while Figures 9-3 and 9-4 illustrate the Stage 1 engine and the Stage 1 POGO control system.

The reliability of Titan derives from the experience gained in the design, fabrication, test, and flight of the entire Titan family (see Figure 9-5). Over 400 Titan vehicles have been built. This number spans the Titan I, Titan II, Gemini, and Titan IIIA, Titan IIIB, Titan IIIC, and Titan IIID configurations. Reliability of the Titan III members of the family was enhanced through the common core concept wherein Stages I and II of all Titan III vehicles are essentially identical. The core for all configurations uses common parts, common drawings, and common manufacturing and testing processes. Modifications to meet specific mission requirements were made with discrete kits that did not degrade the reliability integrity of the basic core. Table 9-1 lists notable firsts.

Table 9-1. TITAN III NOTABLE FIRSTS

Flight	Mission	Launch Site	Date
1	First Titan III-C Flight	Cape Canaveral	06/18/65
6	Unmanned Gemini Capsule	Cape Canaveral	11/03/66
16	Improved TVC	Cape Canaveral	05/05/71
17	First Titan III-D Flight	Vandenberg Air Force Base	06/05/71
25	Silver Anniversary Flight	Vandenberg Air Force Base	07/13/73
28	First Titan III-E Flight	Cape Canaveral	02/11/74
32	First Helios Flight	Cape Canaveral	12/10/74
35	Viking II Mars	Cape Canaveral	08/20/75
36	Viking I Mars	Cape Canaveral	09/09/75
39	Second Helios Flight	Cape Canaveral	01/15/76
47	Voyager 2	Cape Canaveral	08/20/77
48	Voyager 1 (final Titan III-E flight)	Cape Canaveral	09/05/77

TITAN ICBM

No information could be found for this study which directly concerned the Titan I verification process.

A single declassified technical operating report titled, Titan II Missile N-1 Flight Test Report, dated September 1962, provided a glimpse of the Titan II verification process. Pertinent excerpts are reprinted here.

> "YLR87-AJ-5 Engine Performance. The steady state engine performance for each subassembly, represented by Pc, was approximately 6 percent low compared to the Pc predicted for the engine at propellant inlet condition for the flight. The cause of the low performance of approximately 6 percent on both engine subassemblies has not been absolutely determined to date. A possible explanation of the low performance would be the presence of approximately 3 to 4 percent water in the fuel propellant. This percentage of water-contaminated fuel propellant could cause a 5 to 6 percent drop in performance as demonstrated by research and development tests conducted at Aerojet-General Corporation. Similar low performance of the YLR87-AJ-5 engine was evidenced on the flight readiness firings which fired for 13.4 sec. At that time, both subassemblies closely indicated the same percentage of low performance. This was attributed to faulty data."
>
> "YLR91-AJ-5 Engine Performance. The engine operated at approximately 50 percent of the predicted level of performance. This caused the missile to fall short of its intended range. The low performance was attributed to leakage or blockage of the oxidizer flow to the gas generator. The abnormality became evident during the start transient and continued throughout the duration of engine operation.
>
> "It would not be possible to distinguish between leakage or blockage downstream of the oxidizer venturi with the instrumentation available.
>
> "The most probable cause of the failure is blockage of the gas generator oxidizer flow within the gas generator manifold by contaminants. These contaminants may have been introduced when the gas generator was retrofitted with an improved seal between the oxidizer check valve and the oxidizer inlet port. This retrofit was incorporated in accordance with Engineering Change Proposal No. 6016 after the gas generator was acceptance-tested."

GEMINI TITAN

The Gemini Program Launch Systems Final Report, Aerospace Report No. TOR-1001(2126-80)-3, dated January 1967, provides insight for this early manned verification process. At the beginning of the Gemini program, the Titan II ballistic

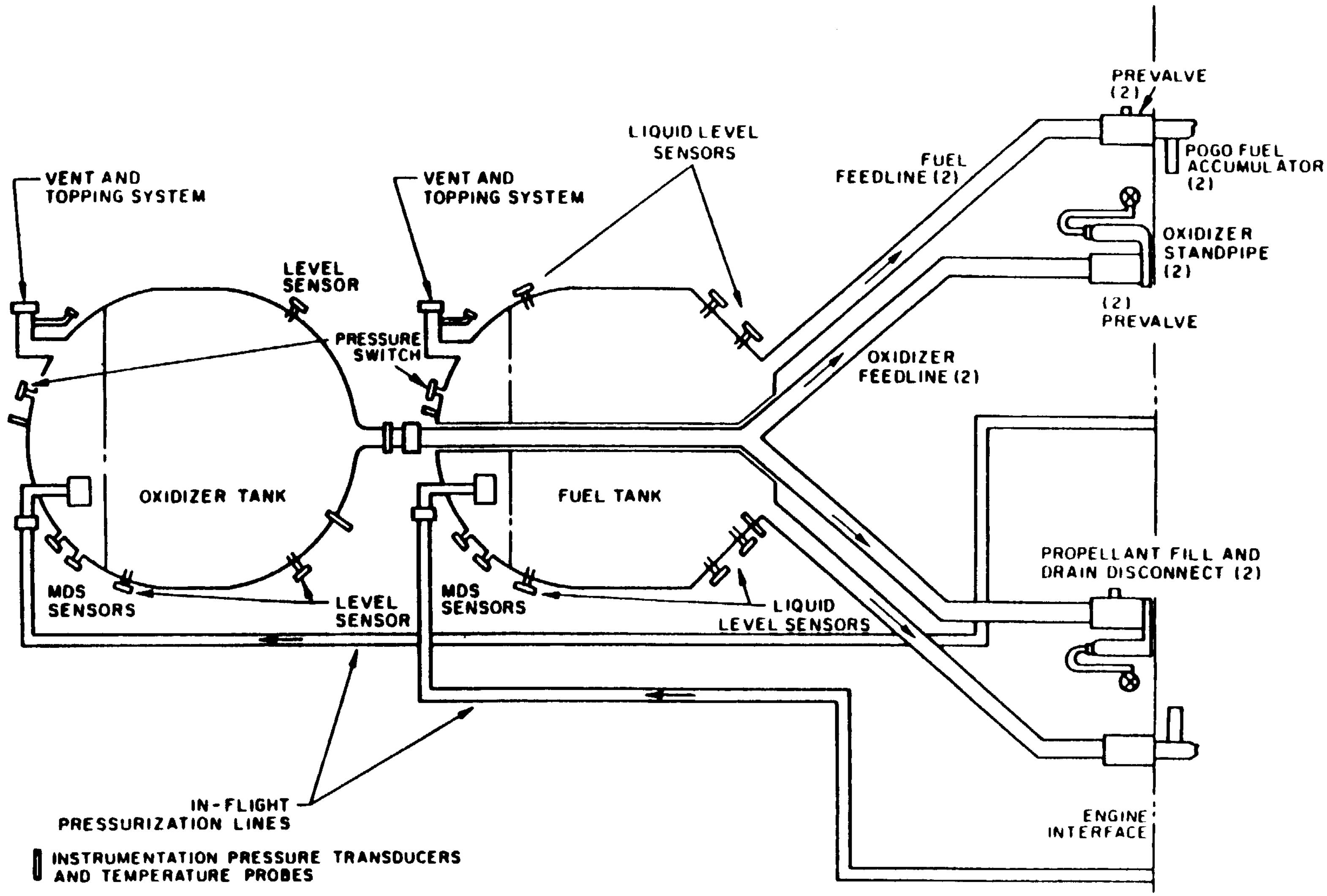

FIGURE 9-1. STAGE I PROPULSION SYSTEM

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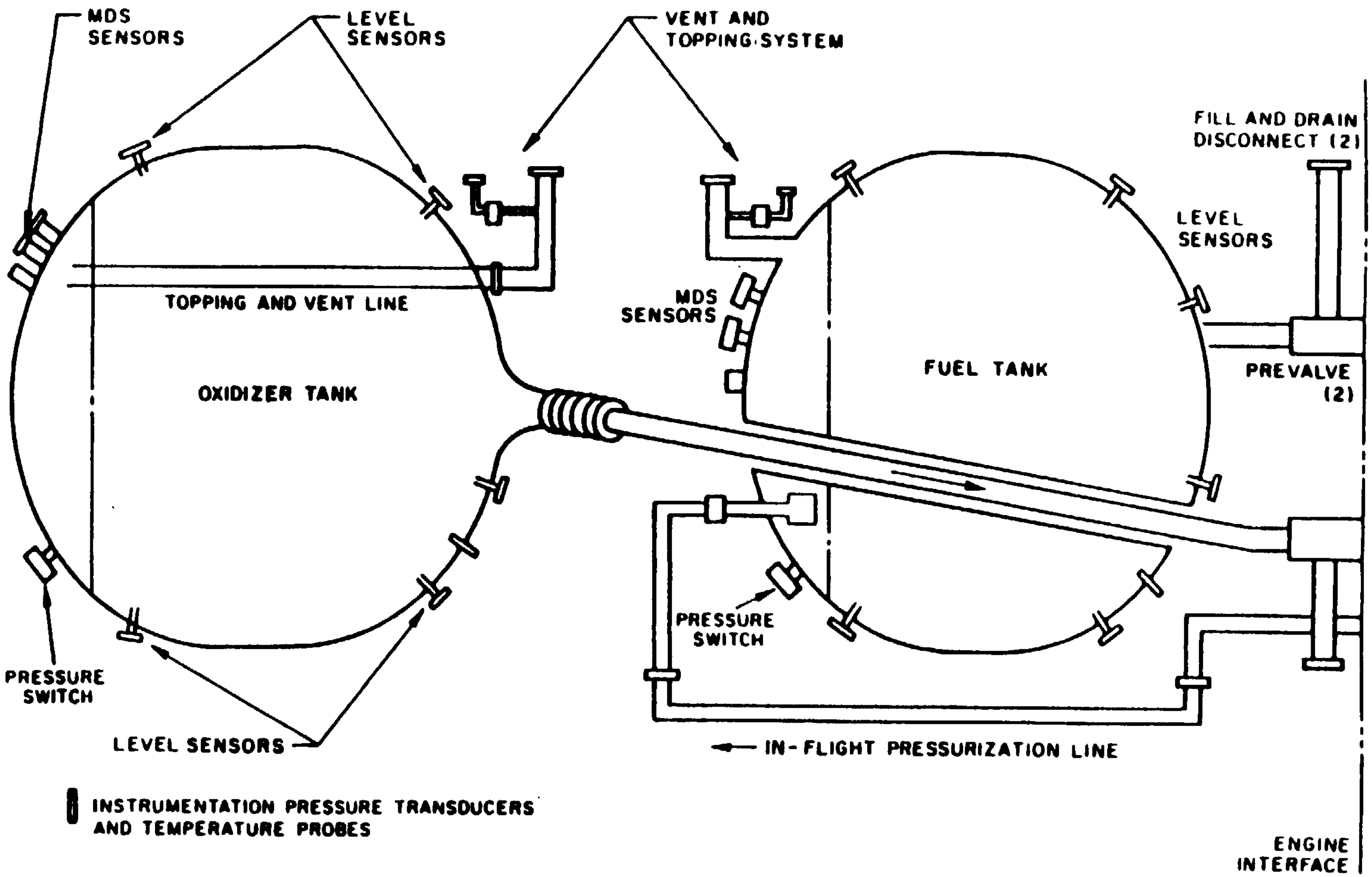

FIGURE 9-2. STAGE II PROPULSION SYSTEM

MDSTCPS - [M]ALFUNCTION DETECTION SYSTEM THRUST CHAMBER PRESSURE SWITCH
OPPS - OXIDIZER PRESSURANT PRESSURE SWITCH
FPDPS - FUEL PRESSURANT DIFFERENTIAL PRESSURE SWITCH
LTCV - POSITION, THRUST CHAMBER VALVE
POT - POTENTIOMETER

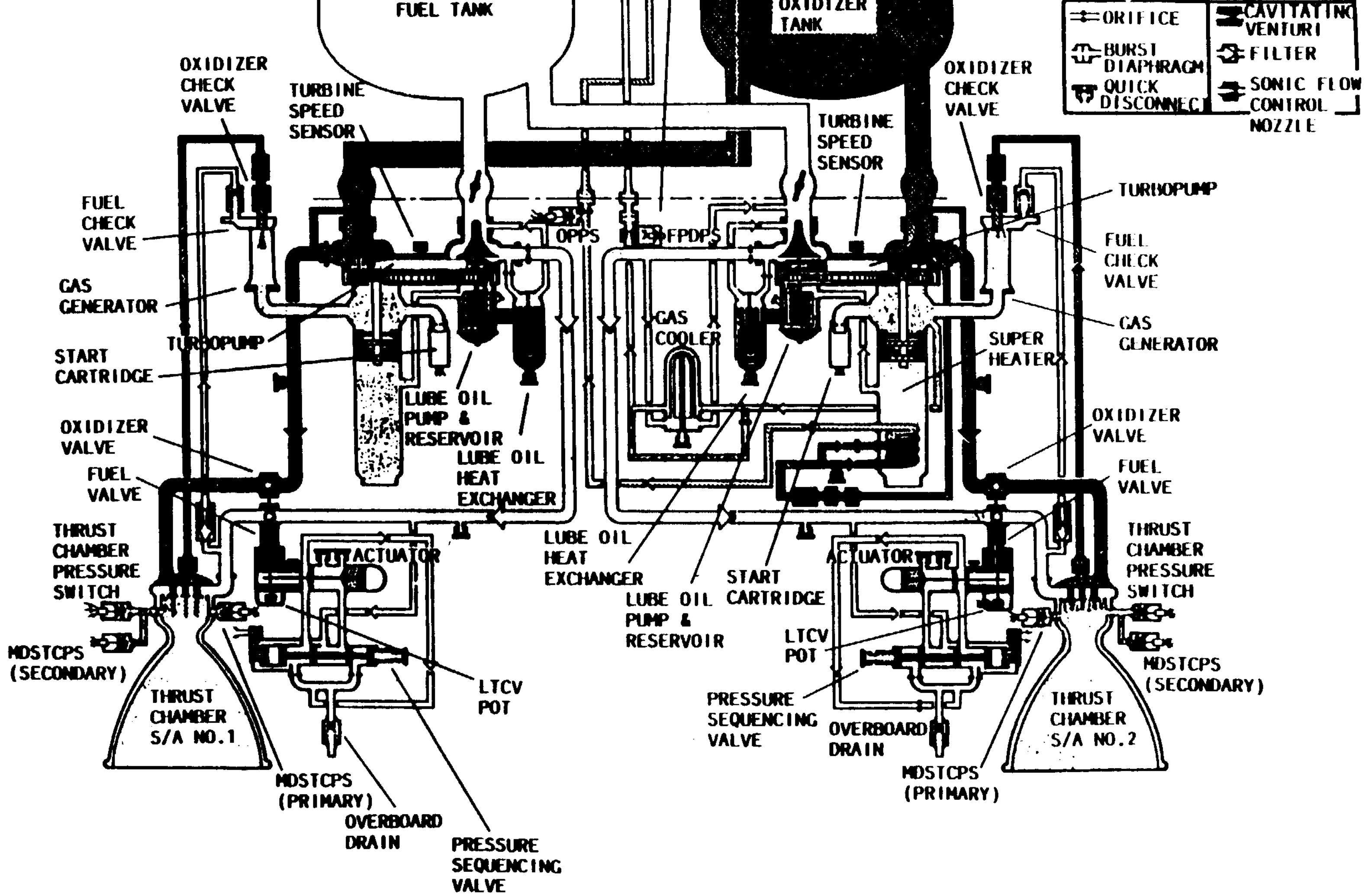

FIGURE 9-3. ENGINE STAGE I

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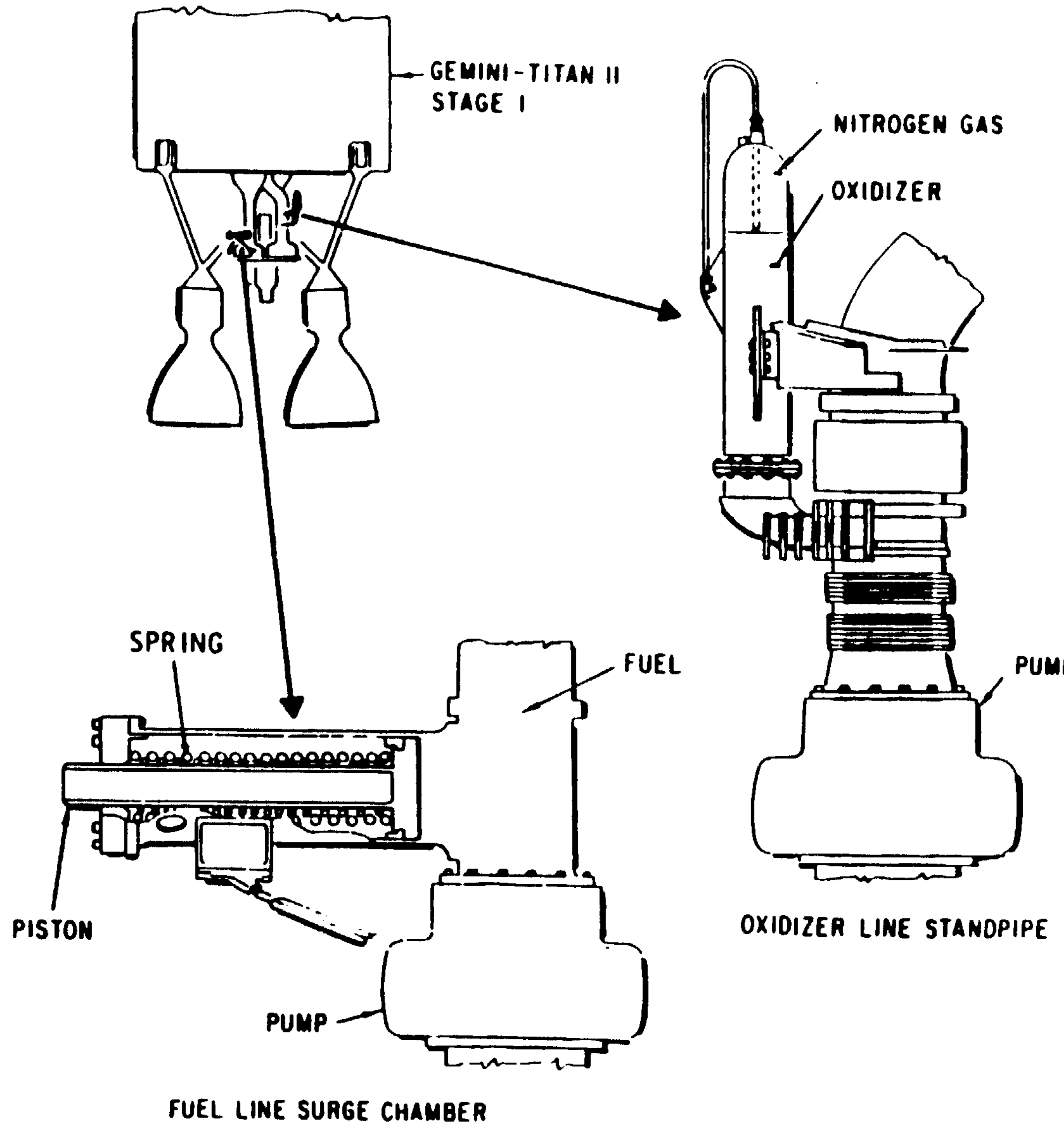

FIGURE 9-4. "POGO" SUPPRESSION EQUIPMENT

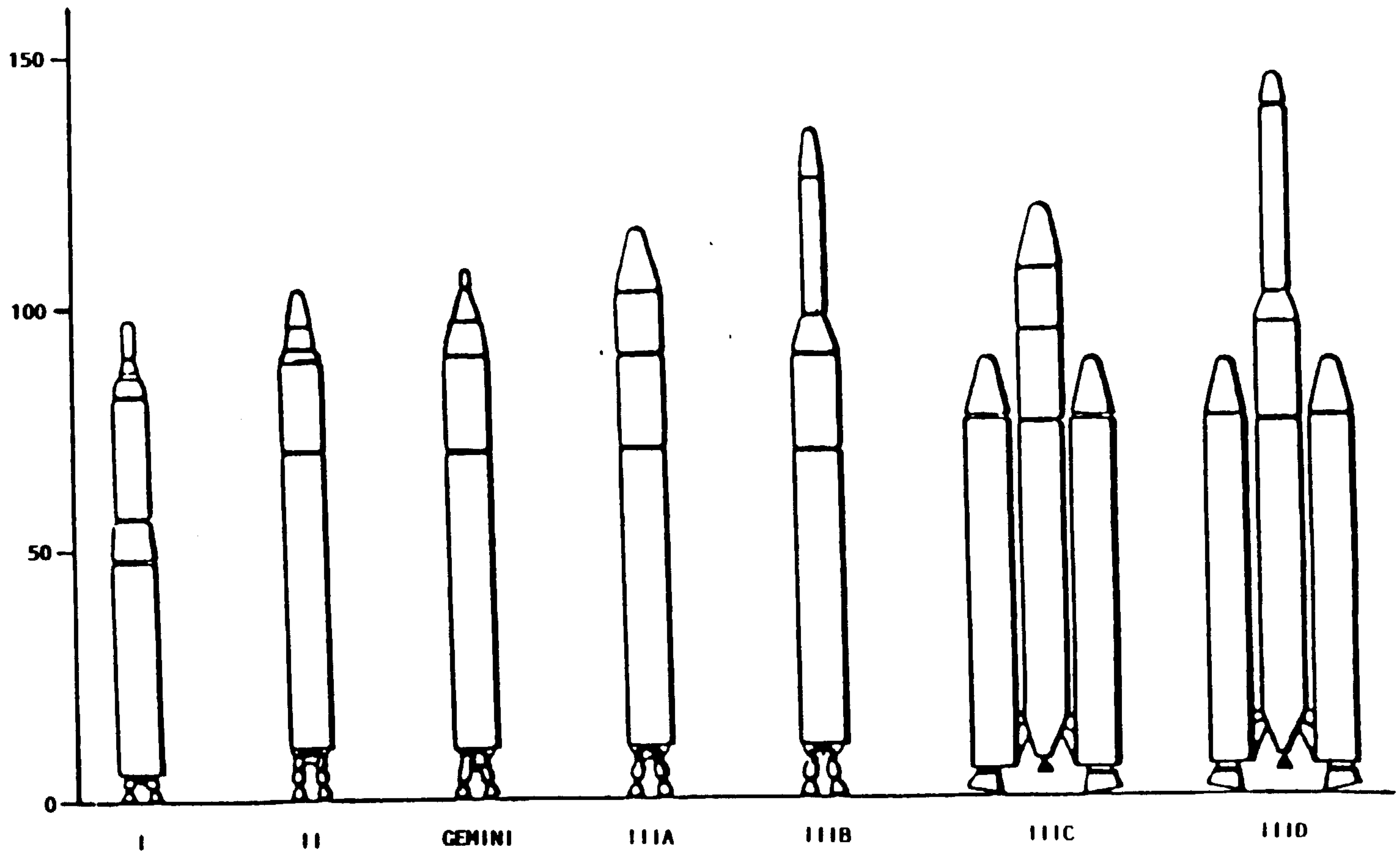

FIGURE 9-5. TITAN LAUNCH VEHICLE FAMILY

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missile was chosen as the launch vehicle to be used in conjunction with the Gemini spacecraft. The decision was made at that time to use the Titan II "as is" with only those modifications required to enable the Titan II to perform the launch vehicle function for a manned system.

The Gemini Launch Vehicle (GLV) engine configuration (-7 model) used the basic Titan II (-5 model) as a building block. Certain changes were made in the basic Titan II engine to adapt it to the critical uses of the Gemini program. The changes were prompted by the need to man-rate the Titan II, and/or by experience gained in the early Gemini flights.

The propulsion system of the GLV was a direct adaptation of the Titan II propulsion system. Some deviations or additions were made to meet specific Gemini requirements. The launch vehicle used the Aerojet-General YLR 87-AJ-7 liquid propellant rocket engine for Stage I and YLR 91-AJ-7 liquid propellant rocket engine for Stage II, respectively. These engines burned storable hypergolic propellants, nitrogen tetroxide and UDMH-hydrazine blend. The propulsion systems included a propellant feed system and a tank pressurization system. The propellant feed systems for first and second stages contained the tanks, feedlines and associated valving necessary to store propellants in the vehicle and supply these propellants to the rocket engines. The first stage propellant system also contained hydraulic oscillation suppression devices necessary to eliminate longitudinal oscillation instabilities (POGO) caused by closed loop coupling between the structural resonances and the propulsion system. The tank pressurization system was used to provide proper propellant pressure to the engines during start and flight.

DEVELOPMENT HISTORY - GEMINI

The Gemini development of the propulsion system was conducted in the propulsion system test program (PSTP) at Aerojet-General, Sacramento, January 28, 1963 to March 2, 1964. The purpose of the test program was to check and verify the operation of that portion of the Titan II/Gemini propulsion system peculiar to the GLV.

The PSTP results demonstrated the ability of the Gemini propulsion system to meet its design requirements. Tests were run to determine whether reduced fuel and oxidizer tank ullage volumes had adverse effects on engine starting characteristics. Results demonstrated that reduced ullages did not cause violation of engine net positive suction head requirements. Validity of the analytical technique to predict pressurization system performance was also satisfactorily demonstrated. The effects and calibration of engine performance with cold propellants were also demonstrated in these tests.

Because of the performance requirements imposed on the launch vehicle, level sensor performance and reliability became significant factors. The flight test failures on Titan II and the problems encountered in attempting to qualify the sensors resulted in a decision to change from the Titan II level sensor to a Bendix unit for Gemini.

"Piggy back testing" was used to fly GLV hardware on Titan II R&D vehicles. Nineteen Bendix level sensors were flown on Titan II Flights N32 and N33. Flight uncovering times were compared with readings from the Titan II level sensors in equivalent tank locations. The agreement between all pairs of sensors was within tolerance. On these flights, the first indications of fuel sensor uncovery and subsequent recovery were noticed. This condition was verified on GT-1, and subsequent corrective action shielded the fuel sensors from pressurization gas condensation.

The first Titan II flights showed a very high "g" level of oscillation in the first longitudinal mode in the time period preceding booster engine cutoff. The "g" levels were too high for the equipment aboard the Titan II, much less a pilot. Titan II pursued the problem and found that an adequate reduction in level for a weapons system could be achieved by simply increasing fuel tank flight pressures. The reduced levels of oscillation resulting from this change were not, however, compatible with a manned requirement. A special program was established to evolve a fix that would reduce the levels to the +.25 requirement for a manned vehicle. The proposed fixes of detuning the feed system, the standpipe and fuel accumulator, plus providing an attenuation of pressure oscillation frequencies in the structural band, were tested and demonstrated successfully in three Titan II flights, N-25, N-29, and N-31. These flights verified the major portion of the initial Gemini redesign of the Titan II experimental hardware. The N-31 flight (flown with reduced fuel tank pressure) also verified that the hardware was effective at a reduced fuel tank pressure level, which was shown to be a strong parameter in controlling the levels of oscillation without the fixes.

Considerable effort was directed toward the development of a remote charging system (AGE) which was used to tune the launch vehicle oxidizer standpipes. The system was developed to provide greater flexibility in mission planning. The original manual charging technique involved opening prevalves and dropping propellant relatively early in the countdown, thus exposing the engine to oxidizer. This exposure necessitated a greater recycle time in the event of a launch scrub. The concept of the remote charging system provided increased safety in the charging operation and greater flexibility when the charging operation was performed because it was controlled from the blockhouse. It was proven through development test and subsequent flight test experience that the remote charge system provided a more consistent charge than the manual system.

QUALIFICATION HISTORY - GEMINI

Components in the GLV, like Titan items, were qualified by similarity to their respective Titan II counterpart. Qualification problem areas existed such as the POGO fuel accumulator position potentiometer, the tank topping system vent disconnect, the tank topping system solenoid valve, POGO airborne ball valve (shutoff valve) and the propellant tank level sensor.

The basic POGO hardware, oxidizer standpipe and fuel accumulator, were qualified by successful flight test on Titan II Flights N-25, 29, and 31. During

component vibration testing of the POGO fuel accumulator rotary potentiometer, the potentiometer mounting screws loosened and the negator spring which connects the potentiometer to the accumulator piston slipped off the rim of its potentiometer attach wheel. Corrective action was taken to strengthen the potentiometer mounting provisions and these failure modes were eliminated. The potentiometer failed the propellant compatibility test due to penetration of oxidizer through the front bearing of the pot. Corrective action consisted of installing an O-ring seal external to the bearing area.

Problems were encountered with the remote operated airborne ball valve in that the ball rotated to a slightly open position under vibration environment. The problem was solved by incorporating a spring detent mechanism to positively lock the ball in the fully closed position.

Excessive connecting load problems were encountered with the tank topping system disconnect during qualification temperature testing. Corrective action included additional lubrication and tightening of valve and seal cleaning requirements. Sand and dust tests resulted in entry of contamination into the disconnect valve. Corrective action added seals to the open areas.

The tank topping system solenoid valve failed to pass qualification vibration tests by exhibiting sporadic leakage. Corrective action consisted of changing the valve design from a normally closed to a normally open valve.

The Bendix level sensors were subjected to a complete qualification test program. No significant problems were encountered with the component during this program.

Using the Titan I and Titan II environmental criteria as a base, qualification tests were conducted on components which were new, which underwent a change in design or usage, or which experienced a more severe GLV environment. The test program required rigid controls and precise documentation. All failures were recorded, and Quality Control assured compliance to approved specifications, so that qualification testing became a formal program.

CONCLUSIONS - GEMINI

The lessons learned from the Gemini qualification test program were many and should be invaluable to a new program. Two significant areas are described below:

1. It is important to separate the evaluation/development testing from formal qualification testing. The evaluation tests should start with prototype hardware, as soon as the level of the critical environments are established.

2. Failures are an integral part of the test program, but no qualification test schedule has ever been permitted a "failure pad," because this would be admitting that an inadequate evaluation test program had been conducted, or

the design group had serious doubts as to the integrity of the components being tested.

No flight failures were recorded on GLV hardware throughout the program, but one significant failure occurred after engine ignition that caused a shutdown on the pad. This failure (a tandem actuator) probably could have been prevented by a better design review or a better, more extensive qualification program.

SPACE TITAN

The Titan III story began in September 1961 with the initiation of the program definition phase for a standardized space launch system. The original concept for Titan III identified a standard launch vehicle that would be capable of delivering multiple payloads of various sizes and shapes into precise orbits or space trajectories. The requirements were also levied for operational simplicity and a reliable launch-on-time capability. This concept stimulated demanding, technological achievements. Problems and flight failures were encountered. Most important, it provided experience and knowledge which contributed to a successful operational manned launch system.

The Titan III family of launch vehicles currently includes Titan IIIB, IIIC, and IIID. The Titan IIIB configuration launched from Vandenberg AFB for classified military missions is composed of the Titan III common core and the Agena upper stage. Titan IIIC consists of the common core with two 120-inch diameter, 5-segment, solid rocket motors (SRMs) for its liftoff stage and Transtage as its upper stage. Strap-on SRMs used as the initial stage of Titan III vehicles are referred to as Stage Zero in order to retain the Stage I and Stage II nomenclature for the common core. Titan IIID is designed for classified military missions also. This configuration used two 5-segment, 120-inch diameter SRMs identical to those of Titan IIIC, in conjunction with the common core, but it has no upper stage.

GEMINI LAUNCH VEHICLE PROPULSION SYSTEM

The propulsion system of the GLV was a direct adaptation of the Titan II propulsion system. Some deviations or additions were made to meet specific Gemini requirements. The launch vehicle used the Aerojet-General YLR 87-AJ-7 liquid propellant rocket engine for Stage I and the YLR 91-AJ-7 liquid propellant rocket engine for Stage II. These engines burned storable hypergolic propellants, nitrogen tetroxide and UDMH-hydrazine blend. The propulsion systems included a propellant feed system and a tank pressurization system. The propellant feed systems for first and second stages contained the tanks, feedlines and associated valving necessary to store propellants in the vehicle and supply these propellants to the rocket engines. The first stage propellant system also contained hydraulic oscillation suppression devices necessary to eliminate longitudinal oscillation instabilities (POGO) caused by closed loop coupling between the structural resonances and the propulsion system. The tank pressurization system was used to provide proper propellant pressure to the engines during start and flight.

Component Configuration

The propellant feed systems were designed to produce a minimum pressure loss at the design flow rates. The Stage I and Stage II propellants were stored in ten-foot diameter aluminum tanks which also formed the primary launch vehicle structure. Oxidizer was fed to the engine turbopump inlets by an aluminum feedline, passing through a conduit in the fuel tank. For Stage I, the oxidizer feedline was divided in the engine compartment to feed each engine subassembly. The fuel tank outlets were located immediately above the engine pump inlets.

The propellant tanks had anti-vortex and anti-slosh baffles. In addition, the oxidizer tank outlets were designed to prevent loss of energy in discharging the propellant from the tanks.

Propellant prevalves were used in the propellant feed system. The prevalves held propellants above the engine pumps and thrust chamber valves until opening was necessary for terminal countdown. The launch vehicle was filled and drained of propellants through manual disconnects located upstream of the prevalves.

Liquid level sensors were installed in the fuel and oxidizer tanks to facilitate propellant flow rate and outage measurement for performance calculation.

Propellant temperature probes were provided in all propellant tanks to ensure that propellant temperature at liftoff did not exceed the allowable limits based on vehicle performance margins for each mission.

A propellant fill-and-drain system was used to transfer propellants from the AGE to the vehicle tanks. The ground system included transfer pumps, and instrumentation to assure that propellants were transferred to the vehicle at proper temperatures and in the proper quantities.

From the initial lock-up condition prior to launch until engine shutdown at burnout, the pressurization system was designed to provide adequate propellant tank ullage pressure to satisfy (a) minimum NPSH requirements for the engine driven propellant pumps and (b) minimum launch and inflight structural requirements of the tanks. Simultaneously, the pressurization system was not supposed to exceed pressures as defined by structural limitations.

Initial pressurization of all propellant tanks was achieved with a charge of nitrogen gas regulated to a predetermined level. In flight, tank pressurization was provided to both fuel tanks by using cooled rocket engine turbine exhaust gases. The Stage I oxidizer tank was pressurized by vaporized oxidizer from the engine. The Stage II oxidizer tank was pressurized in flight by the initial nitrogen lockup pressure and supplemented by vaporization of the tanked oxidizer. Burst discs were installed in each tank pressurization line.

The inflight pressurization subsystem for Gemini was essentially the same as that used in Titan II. Exceptions were as follows: (a) The pressurant flow to the Stage I fuel tank was increased by a change in the engine flow control orifice size,

and (b) Stage I oxidizer pressurant flow rate was increased through an orifice change and the tank "lockup" pressure was increased.

A revised tank pressure control system known as the tank pressure topping system was designed for the GLV. This system incorporated a pressure supply and vent line which was in parallel with the main pressure supply line available to each tank. When the main lines were disconnected, pressure control was retained through the smaller pressure topping lines which remained connected until liftoff. Lanyard operated pull away couplings were used to separate the GLV from the AGE and effect a seal at the vehicle skin line. An airborne, solenoid-operated,, normally open valve was located upstream of the disconnect coupling on the GLV. This valve provided seal redundancy in flight and a method for leak testing during the prelaunch countdown.

One of the big differences between Titan II and GLV in the propulsion system was the use of suppression devices to control the longitudinal oscillation that occurs during first stage flight. The POGO suppression devices consisted of tuned resonators inserted into the engine feedlines just upstream of the first stage pumps. These devices were tuned to provide attenuation of pressure oscillations in the frequency band of the structural first longitudinal mode. In the oxidizer lines the resonator was a standpipe, while in the fuel lines a tuned mechanical accumulator was used.

The oxidizer standpipe worked on the principle of using an entrapped gas bubble to provide a soft spring for the oxidizer mass in the standpipe to act upon. The energy due to pressure oscillations could be transferred to this spring mass system at the desired frequency by proper choice of the volume of the entrapped gas bubble. The fuel surge chamber or accumulator used a helical spring and piston to perform the same job. The spring together with the piston and fuel mass in the accumulator acted to provide the desired resonance.

GLV ENGINES

Propulsion for the GLV system was provided by the Aerojet-General Corporation manufactured YLR87-AJ-7 Stage I and YLR91-AJ-7 Stage II engines. These engines were modified Titan II engines.

The basic Gemini engine system and those major features which were unique to the Gemini system are described below.

Stage I Engine

The YLR87-AJ-7 rocket engine assembly was a storable liquid bi-propellant, turbopump fed, dry-jacket-start engine rated for sea level operation. The engine was composed of two independently operating subassemblies which operated simultaneously, mounted on a single thrust structure. Each subassembly contained a thrust chamber assembly, turbopump assembly, gas generator assembly, starter cartridge, propellant plumbing, and electrical controls harness. In addition, subassembly two (S/A 2) provided the energy source for tank

pressurization. The major components of each subassembly were identical except for the pressurization system.

The design concept of this engine system was to achieve reliability through simplicity. This goal was achieved by use of hypergolic storable propellants which eliminated the need for bleed down, heaters and ignition systems. The engine required no functions during countdown beyond admission of propellant to the engine from the tankage. Thrust level was preset by sized cavitating venturis which were located in the gas generator propellant feed circuits. Cavitating venturis controlled flow to the gas generator which controlled turbine power by hydraulic equilibrium, thereby eliminating all servo thrust control mechanisms.

Descriptions of the major components follow.

Thrust Chamber Assembly. The thrust chamber assembly consisted of a regeneratively fuel cooled tubular construction combustion chamber, injector, injector dome, fuel thrust chamber valve assembly, oxidizer thrust chamber valve assembly, gimbal assembly, thrust chamber pressure switch, and oxidizer and fuel propellant lines. The above components were packaged as an integral assembly.

The thrust chamber fuel valve consisted of a four inch gate butterfly valve with attached hydraulic actuator. The actuator was spring loaded in the closed position. The actuator shaft was mechanically linked to the fuel valve shaft. The fuel valve shaft in turn had a clevis for linkage to the oxidizer thrust chamber valve.

Control of the thrust chamber fuel valve actuator was achieved by use of a direct mounted pressure sequence valve (PSV). The PSV mechanically sensed fuel system supply pressure and opened at a level which allowed fuel pressure to overcome the spring force of the thrust chamber fuel valve actuator and open the fuel and oxidizer valve by mechanical linkage. Closure of the thrust chamber valve was achieved by an electrical solenoid which overrode the pressure sensing element. This caused the PSV to neutralize and admit fuel pressure to the closing side of the thrust chamber actuator to assist spring closure of the thrust chamber valve.

Gas Generator Assembly. Each gas generator system consisted of an integral gas generator chamber and injector, fuel check valve, oxidizer check valve, gas generator feed lines, cavitating venturis and propellant strainers. The gas generators were mounted directly to the inlet of the turbine pump assembly turbine drive manifold.

Turbopump Assembly. The turbopump assembly consisted of a fuel pump subassembly, oxidizer pump subassembly, two-stage turbine drive assembly and a gearbox assembly. The gearbox contained an integral oil pump, reservoir filtration system and lube oil cooler. These components were integrated into an assembly on the basic gearbox structure. The Gemini gearbox incorporated gears of 9310 alloy and SKF bearings exclusively.

Solid Start Cartridge. The solid start cartridge was comprised of a steel combustion chamber which housed two concentric cylinders of ammonium nitrate propellant (AMR 2506). An ignition train assembly was mounted at the head end of the cartridge. Burning rate of the cartridge was controlled by a flow control nozzle, as well as by stringent maintenance of start cartridge temperature, through the use of cartridge thermocouples and temperature controlled air supplied to the external surfaces of the cartridge. A teflon burst diaphragm over the nozzle hermetically sealed the cartridge assembly. A pyrotechnic squib was installed in the cartridge and electrically fired to initiate the ignition train.

High Pressure Propellant Lines. The high pressure fuel and oxidizer lines consisted of bolted flanges, tubing, and three 2-axis articulated flex joints. The flex couplings were arranged to allow for assembly tolerances and to allow thrust chamber flight control flexure obviating the need of rotary seals to achieve thrust chamber motion. All thrust chamber gimbal motion was taken between the chamber assembly and turbopump assembly. The turbopump assemblies were stationary to the frame and did not move during gimbal operation.

Low Pressure Propellant Plumbing. The fuel and oxidizer lines connecting the turbopump to the tank outlets consisted of low pressure bellows assemblies, modified by POGO gear, from tank outlet to pump inlet. The bellows assemblies were for the purpose of absorbing relative deflections and to allow small misalignment of assemblies.

Subassembly Two (S/A-2) Oxidizer Pressurization Super Heater Assembly. The super heater assembly was a cross flow tube-type heat exchanger located in the turbine exhaust in the subassembly two turbopump. Energy in the turbine drive exhaust gases was used to vaporize and disassociate liquid nitrogen tetroxide. The amount of pressurant gas was controlled by a cavitating venturi into the liquid side of the circuit. Energy in the gas was controlled by a back pressure nozzle in the gaseous side of the circuit. The oxidizer pressurant system included the associated plumbing.

Subassembly Two (S/A-2 Fuel Tank Pressurant Heat Exchanger. The gas cooler assembly consisted of a bundle of U-shaped tubes enclosed in a cylindrical shell. Fuel was circulated on the interior side of the tubes as a heat exchange medium. Hot gas passed through the external passages around the tubing to the fuel autogenous system. The hot gas flow rate was controlled by a sonic nozzle in the plumbing circuit.

Sequence of Operation. At a minimum of 30 seconds prior to engine start, propellant valves at the tank outlets were opened allowing propellant into the engines up to the thrust chamber valves (TCV's). The engines were self-bleeding by gravity head.

Each Stage I subassembly start was initiated by providing a 28-vdc signal to a redundant bridge wire squib in the solid start cartridge initiator. The initiator fired the ignition train which started burning of the solid propellant grain. Energy from the solid propellant caused acceleration of the turbopump assembly. As the

turbopump assembly generated fuel pressure head, the pressure sequence valve shuttled at a pre-set pressure causing the thrust chamber fuel valve to be actuated open by fuel system pressure. The oxidizer valve was opened by mechanical linkage to the thrust chamber fuel valve. Propellant flow from the turbopump filled the volume in the thrust chamber assembly and ignition occurred upon contact of the propellants in the combustion chamber. As a result of thrust chamber combustion backpressure, propellant was supplied through gas generator feed lines downstream of the thrust chamber valve initiating the regenerative (bootstrap) cycle. As a result of the initiation of bootstrapping, the engine achieved rated thrust. Energy of the start cartridge was controlled to burn out immediately after the start of the bootstrap operation as the engine approached steady state operation. The thrust level was controlled by tuning pre-sized cavitating venturis to achieve hydraulic equilibrium. No electrical power whatever was required by the engine during steady state operation.

During these initial moments of life of the engine, two Pilot Safety systems were monitored for engine parameters/characteristics in order to detect any abnormal operation. The first was known as the Prelaunch Malfunction Detection System and was operative until the vehicle was airborne. The second, known as the Malfunction Detection System, continued operation and surveillance throughout Stage I flight.

The PMDS was designed to monitor the Stage I autogenous system operation prior to release of the launch vehicle and to furnish go/no-go signals to launch control equipment in response to the conditions sensed in the fuel and oxidizer autogenous systems.

The MDS was a warning system that provided a visual cue to the astronauts in the spacecraft in the event of a launch vehicle subsystem malfunction which could possibly result in the failure of the flight mission. The Malfunction Detection Thrust Chamber Pressure Switch (MDTCPS), a part of the MDS package, also provided a signal to launch control equipment verifying attainment of proper thrust prior to vehicle release, as well as serving to initiate the controlled shutdown sequence whenever a pre-determined decay in thrust chamber pressure had occurred.

Engine shutdown was accomplished by either oxidizer exhaustion or fuel depletion. Although neither type of shutdown caused problems on the Gemini engines, the oxidizer exhausting was more desirable from a hardware integrity standpoint and shall be used to further describe the shutdown sequence. As oxidizer was depleted, the MDTCPS sensed a decay in thrust chamber pressure; the switch opened, causing a 28 vdc signal to be sent to the override solenoid of the PSV. The PSV override shuttled the PSV, causing fuel system pressure to be vented from the opening side and applied to the closing side of the actuator. As the thrust chamber valves closed, the propellant supply to the gas generator was terminated with a resulting thrust decay of each subassembly.

Major Changes. As previously noted, the GLV engine configuration (-7 model) used the basic Titan II (-5 model) as a building block. Certain changes were made in the basic Titan II engine to adapt it to the critical uses of the Gemini

Program. The changes were prompted by the need to man-rate the Titan II, and/or by experience gained in the early Gemini flights. Although many changes were incorporated during the Gemini Program, certain major changes warrant sufficient attention to be singled out for special mention.

Stage I Gearbox. Failure of the idler gear which resulted in catastrophic failure of the turbopump was encountered. Strengthened gears were incorporated on all Stage I gearboxes for GLV-2 through 12. A second failure, which required resolution prior to manned flight, centered around failure of the number 6 bearing. Gemini's solution prescribed use of SKF bearings only and redesign of the turbine interchange labyrinth seal.

Stage I Engine Frame. The Stage I engine frame was qualified by similarity to the Titan II engine frame. However, because of certain, though minimal, modifications the capability of the Gemini engine frame under the conditions of firing and gimballing was verified during the propulsion system test program.

Flexible Lube Oil Cooler Coolant Lines. The fuel return line (from the oil cooler) fractured approximately 30 seconds after the start of a hot-fire test on Gemini engine GLV-1011. The failure was attributed to excessive loads due to vibration and assembly distortion. A change to flexible inlet and outlet lines (for the fuel coolant) was incorporated to preclude further occurrences of this nature.

Start Cartridges. Two problems were encountered which were resolved by incorporating non-interchangeable (between Stage I and Stage II) diaphragms and temperature conditioning of the cartridge itself.

Propulsion System Test Program. This program was promoted to evaluate/demonstrate the satisfactory operation of Gemini unique components and requirements for the Stage I and Stage II propulsion systems.

0-5 Volt Instrumentation System. The Gemini program used a 5 volt full scale telemetry system, whereas Titan II had used a 40 mv. full scale telemetry system. The higher output voltages from remote transducers on the Gemini system improved the overall signal-to-noise ratio and reduced the complexity of the multiplex/encoder.

TCV Stress Corrosion. Failure of a thrust chamber fuel valve body prior to liftoff on one of the Titan II vehicles was attributed to stress corrosion. Investigation resulted in a change of heat treat from T-6 to T-73 on Gemini.

Stage II Engine

System Description. The Stage II engine system was in general identical in design concept to that of the Stage I engine system. The distinguishing features which differentiate the Stage II engine from the Stage I engine were the following:

1. Stage II engine components were a reduced thrust version of Stage I subassembly.

2. Rated for altitude start and operation with a baffled injector.
3. The high expansion ratio chamber (49.2:1) was achieved by use of an ablative extension from 13:1 to 49.2:1.
4. Turbine exhaust gases were ducted through a swiveled nozzle assembly to provide roll control during Stage II powered flight.
5. Pressurization as required only on the fuel tank of the Stage II vehicle. Oxidizer tank pressurization was supplied by precharge and boiloff only.
6. The Stage II cartridge incorporated the same design features as the Stage I except that the propellant was a single, longer cylindrical section. It also included a conditioning system similar to Stage I.
7. Stage II had no thrust chamber pressure switch (TCPS).
8. Stage II incorporated a redundant shutdown system (RESS) which is described later in this report.
9. Stage II did not have a PMDS system and the MDS switch monitored fuel injection pressure (MDFJPS) not chamber pressure as on Stage I. The MDFJPS was not a part of the shutdown circuitry as the Stage I MDTCPS.

System Operation. The Stage II engine sequence was identical to the Stage I engine subassembly. The same signal which shut down the first stage signaled the second stage ignition. Staging was accomplished by the "fire-in-the-hole" concept. Stage II thrust buildup caused separation of the vehicle stages. As will be noted further on, Gemini had a unique feature of redundant shutdown capability on the Stage II engine. This redundant shutdown capability was achieved by firing a squib actuated valve in the gas generator oxidizer feed circuit simultaneous with the signal to the thrust chamber valve.

Major Changes. This section reports on major components or systems which were unique to the GLV Stage II engine compared to the Titan II Stage II engine. They were also unique to GLV Stage II compared to GLV Stage I unless otherwise noted. These systems were developed and incorporated due to Gemini program requirements or as the result of problem solutions.

Redundant Engine Shutdown System (RESS). A redundant engine shutdown capability was added to minimize the possibility of spacecraft overspeed due to a failure of the basic thrust chamber valve/pressure sequence valve. This redundant system was developed under the augmented engine improvement program and consisted of a squib actuated valve in the oxidizer bootstrap line which was activated by the same signal sent to the PSV. This system was incorporated on GLV-3 and subsequent vehicles.

Fuel Injection Pressure Switch. The malfunction detection system pressure switch initially monitored chamber pressure as on the Stage I engine. However, the initial Titan III Stage II engine which used a similar configuration switch encountered combustion instabilities on two acceptance tests. The analysis indicated the most probable triggering mechanism was detonation in the sensing tube, and the magnitude a function of line volume. As GLV corrective action, the MDS switch was re-located to sense fuel injection pressure, which reduced sensing tube volume to a minimum with only a chamber pressure transducer on the combustion chamber. This action was taken because the standard Titan family

injector (which had a higher susceptibility to instabilities) was on the first seven GLVs. The relocation was effective on GLV-2.

GEMSIP Injector. A dynamically stable injector was developed to increase the combustion stability margin. The new injector, GEMSIP, was incorporated on GLV-8 and subsequent vehicles.

Start Cartridge Temperature Conditioning. As previously described, a solid start cartridge conditioning system was also incorporated on the Stage II engine system.

0-5 Volt Instrumentation System. The 5-volt flight instrumentation system was also incorporated on the Stage II engine system.

The same configuration ablative skirt was utilized of the GLV engine but due to longer burn durations, a program was conducted to demonstrate the capability to withstand these durations. To ensure that the incorporation of the GEMSIP injector was also compatible with these long durations, additional testing was conducted.

APPENDIX 10

STATIC FIRING SATURN STAGES

This MSFC Test Division memorandum (report) is included both for historical data completeness and because it addresses a related issue.

The only copy obtained was several reproduction generations old. To ensure readability, the liberty of re-typing the report and re-working the figures was taken.

Some conclusions drawn by this report are applicable to the value of propulsion system testing.

GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

Memorandum

TO Addressees

DATE December 6, 1968

FROM Chief, Systems Test Division, R-TEST-S

In reply refer to: R-TEST-S-#59-68

SUBJECT Static Firing Saturn Stages

1. In regards to recent conversations on static firing, or not static firing, the data we gave to Dr. Mrazek for his report of February 1968, have been up-dated and some additional evaluation made of the data.

2. These up-dated results are attached for your information.

Daniel H. Driscoll Jr.
Daniel H. Driscoll, Jr.

Enc:
a/s

Addressees:
DEP-T, Dr. Rees/Mr. Neubert
I-MT-MGR, Mr. Balch
I-V-MGR, Mr. Godfrey
I-V-MGR, Col. James
I-DIR, Gen. O'Connor
I-DIR, Dr. Mrazek
I-I/IB-MGR, Col. Tier
R-DIR, Mr. Weidner
R-SE-DIR, Mr. Richard
R-QUAL-DIR, Mr. Grau
R-P&VE-DIR, Dr. Lucas
R-TEST-DIR, Mr. Heimburg/Tessmann
R-TEST-S, File

MSFC - Form 488 (August 1960)

December 4, 1968
Retyped:
May 15, 1989

STATIC FIRING SATURN STAGES

1. INTRODUCTION

The question has often arisen as to the justification for continuing static firing Saturn stages. The question has been asked by program management which seeks a "mathematical" answer. A "mathematical" answer would understandably relieve management of having to make a decision based only on engineering judgement or tradition. The motive is understandable because of the real cost of the present day approach and the fear of a human failure which can lead to the destruction of equipment as it has in the S-IV and S-II.

Any statement regarding static firing of flight stages must be prefaced with a statement as to why flight stages are static fired. Flight stages are static fired to determine the quality (design and hardware) of that particular assembly at a time and place where a flight catastrophic type failure can be best withstood and controlled. To provide this benefit to the program, management must assume that the testing discipline and philosophy is such that it can provide efficient screening of defective design and hardware. To provide the most confidence, management must also assume that the article tested is as nearly as possible the same hardware as that which will subsequently fly. The opportunity to exercise this screening process is not a product of inventive genius, but the simple recognition that people are fallible especially where the results of their errors are not immediately apparent to them as in factory assembly. Management should also recognize that static firing, as opposed to checkout, provides a more realistic test stress and the only real end-to-end check of many of the stage systems and real verification of their functional capability. The objective data gained from this testing is a major, clear, forcing function in the development cycle not only of the hardware but of the people (designer and manufacturer). That neither of these appreciate this surveillance is humanly understandable, but should not be determining factor in its existence. Neither should management yield to their own discomfort when this objective event reveals problems which by hindsight are ridiculous, but whose discovery are the very reason for testing.

Static firing is a natural attribute of liquid propulsion systems (provided their design is not too sophisticated) because the real flight hardware system can be meaningfully, functionally tested on the ground where the static fired, hardware is available for post-test visual inspection and where more detailed and more accurate measurement of performance parameters can be obtained that can in flight.

In this day and age, the design of components of a flight system is relatively amenable to straight forward mathematical analysis where only that component's self contained operational environments are considered. It is

another matter when this component is required to function properly in a system which produces a combination of environments which can only be determined by empirical means. Additionally, these system environments may change from assembly-to-assembly because no two assemblies are exactly the same even when made from the same drawing. Processing freedom (a necessity if you ever want to get a product) and material non-homogeneity account for some of the natural variances from assembly-to-assembly. Therefore, management must assume some variances in a system even though it be tightly controlled by paper.

The obvious, underlying management assumption must be that the test operations can be performed with a minimum chance of catastrophic loss of the test item by human error. Management should therefore allow every reasonable means to be taken to sense and prevent catastrophic loss of proper add-on instrumentation, fire systems, leak detectors, automatic safeing features, and hazards and damage control apparatus. Secondly, extensive instrumentation is available for static firing to be able to determine cause of failure in an objective manner.

The mere existence of a stage static firing capability (facility and crew) provides a key resource to fall back upon for the solution of flight problems and the capability to verify major functional design changes.

Some specific stage systems can only be functionally tested by static firing the stage, (such as active propellant utilization, propellant pressurizationi, engine gimbal actuation under main hydraulic pump power and flight control systems). Conversely, there are some environmental conditions such as vacuum and aerodynamic loads and vibrations which are not economically feasible to simulate on the test stand. Additionally, a large proportion of the flight measuring system for vibration, pressures and temperatures gets its only real end-to-end check through propellant loading and/or static firing.

In payload critical missions, it is very important to have real characteristic data for the specific flight hardware. Such characteristics as the specific propellant tank calibration, depletion efficiency, pressurization performance, and engine performance become most critical to furnish data for flight prediction, especially where no active propellant utilization system is used. Without this real data, statistically sound, and performance trade-off negotations become a matter of pure subjective judgement.

Provided no major functional design changes occur and provided that system quality can be infallibly controlled (both avowed developmental goals) then the gain in confidence provided by static firing should decrease with increasing number of stages produced. It should be borne in mind, however, that there are some characteristics of the Saturn system that work contrary to these goals. Mission requirements have been and probably will be changeable. Many parts are involved (as one contractor proudly claims more than 2 million per Saturn V vehicle) making control very difficult especially when prime contractors do not exercise contractural control of <u>all</u> changes in their vendor's, fixed price procured, components. None the less, proof that not much more is to be gained y continuing static firing should, for any reason, be objectively evident from the decreasing rate of occurrence of critical failures. Two sources of historical information were

evaluated to determine whether static firing had, in fact, provided an objective demonstration of hardware design verification, quality assurance and subsequent successful flight assurance. The static firing and flight histories of non-Saturn programs were evaluated because of the large samples provided and the different approaches involved. Secondly, the results of static firing and flying the Saturn flight stages were evaluated because they are directly applicable.

2. BASIS FOR SELECTION OF REPORTED FAILURES ON NON-SATURN LAUNCH VEHCILES

Data was provided by the contractors themselves and from a report prepared by an outside system contractor. Where in conflict the responsible contractor's data was used. This data was then screened for failures in flight which were caused by systems that were judged to be static firing, testable, and therefore, ideally could have been prevented from happening in flight by a proper static firing test program. The depth of the data did not include what in addition was discovered and corrected in those static firings which were made nor what design changes and, disassemblies were made after static firing and prior to launch. The data provided for these systems can only be used to answer the gross question of whether a learning process did exist and whether static firing, where performed, appeared to allow an acceleration of this learning process. Additionally, the answer might be provided to the simple question of whether static firing, testable systems ever came to such a maturity that failures no longer occurred even when large numbers of flights were made.

3. BASIS FOR SELECTION OF REPORTED DATA ON SATURN FLIGHT STAGES

The critical failures reported herein were selected to the exclusion of other findings because they fitted the situation where they would not have been found by checkouts and therefore, would have occurred during the launch countdown or flight if they had not occurred during the first static firing countdown of that specific stage. These failures were classified into two categories:

a. LAUNCH ABORTS AVOIDED

These were S-1, S-1B and S-1C booster stage systems failures which occurred during the first attempt or accomplishment which would have caused an aborted launch if this operation was assumed to have occurred at the launch site rather than on the test stand. Within this category falls those failures which actually terminated the static firing between ignition and launch commit and those failures encountered which at the launch site during terminal countdown would have required detanking to fix. The case assumes a Wet Countdown Demonstration Test would have eliminated those problems found during propellant load tests and therefore these were not included in this category.

b. MISSION FAILURE AVOIDED

Those booster and upper stage systems' failures encountered during the first static firing attempt or accomplishment which would have in most probability caused a major mission loss if this operation had really

been the flight instead of static firing. Second or subsequent failures were specifically excluded by this screening in order to keep the model mathematically pure.

Saturn static firing experience provided another category of data which is:

OTHER CRITICAL FINDINGS

Those booster and upper stage systems failures which were experienced at any time in the propellant loading and static firing operations but excluding those listed under the first two categories.

UNSATISFACTORY CONDITIONS

(Defects, PARS, FARS, etc.), during the static firing and post-static operation were specifically excluded from this analysis. Some of these conditions were found simply because a new set of evaluators worked with the hardware in a more nearly real functional environment than can safely be provided by factory checkout. These experiences would have surely been passed on to the launch site and caused excessive delays there where the resources for their solution should intentionally be limited so that maximum attention can be focused on the business of launching. Others, however, were unecessarily caused by this additional operation and traffic with the stage. The specific significance becomes a matter of strong subjective judgement and leads to an endless controversy.

4. RESULTS

The data from ten different systems was compiled and evaluated, (2 Army, 3 NASA and 5 Air Force). All except the Gemini, static fired flight stages prior to launch (See TABLE 1). Here the similarity ends.

Those which did employ static firing did not employ a uniform discipline, for instance, short (less than 50% full) duration versus full duration.

Even those which did employ full duration (i.e., Redstone and Jupiter) had less than the full stage system (TM and electrical control) on for static firing unlike the way the Saturn stages are static fired as complete stages. It was unofficially reported that occasionally, Titans were static fired with "slave" engines which were removed prior to shipment to the launch site where the flight engines were installed.

The number of stages which were static fired prior to flight varied from two (2) to fifty-seven (57) before static firing was stopped. In all cases except Titan III (Table 3), the first group of stages was static fired.

The design complexity of the systems varied widely ranging from single stage to three stage (all liquid and combination of liquid and solid). The Titan III and the Saturn 1/1B are not that uncommon if we consider that 1/1B has solid retro rockets and storable APS systems which must function reliably for mission success and which are not static fired prior to launch.

Not all programs were executed concurrently, therefore, the later ones had the opportunity to learn from the earlier.

In the non-NASA programs there was a wide divergence in average yearly launch rate (TABLE 2), from nine (9) to thirty-one (31).

The mission losses (TABLE 2) because of systems testable by static firing was, in all cases, far less than those from all causes, comprising only from 10 to 33%. Of those which did fail because of systems testable by static firing, failures occurred on flight stages which had been static fired prior to launch as well as on flight stages which had not been static fired prior to launch.

Because some learning curve should result, either from static firing and launch or from just launching, a plot was made assuming all selective data points to have the same weight, but still take into account only those mission loss, failures experienced in systems testable by firing. This data is shown plotted in FIGURE 1. There was obvious learning, on this specific basis, in all cases except Atlas. The pay-off did not become apparent on the average until after 25 to 30 launches. However, even after that, failures did persists, but at a much lower frequency.

In the comparison of the fourteen (14) Gemini flights, which was the only system with a complete absence of any kind of static firing, with the first fourteen (14) flights of those systems which had some kind of a static firing program, it can be seen from TABLES 2 and 3 that the SATURN 1/1B, Titan II and Titan III experienced none of these specific type failures, while the Gemini experienced two (2). However, the Redstone, Jupiter, Thor, Atlas and Titan I did experience anywhere from one (1) to six (6). It

should also be noted however, that the Gemini should have drawn heavily on the learning paid for in the preceding flights and tests of the Titan II.

In analyzing the Saturn stage, data contained in TABLES 4 through 8, it can be seen that critical failures were still being experienced from static firing of the S-1B stage even on the eighteenth (18) stage to be static fired.

Learning has been more rapidly achieved on those stages than was apparent from the non-NASA programs. Flight results of Saturn bear out this observation.

The same randomness of occurrence versus sequential order as observed in non-NASA flight results, appears in the Saturn static firing experiences but does not appear in the flight experience.

5. CONCLUSIONS

There is no evidence from these statistics which would give an unqualified answer as to the serial number at which static firing can be stopped with the positive assurance that no failures can occur which could have been avoided by a properly performed static firing.

The tenacity with which failures persist suggests that even under our most extensive (and expensive) work control system learning is limited by other factors.

There is no doubt that static firing of Saturn stages to-date has provided a major contribution to its successful flight history. If the failures listed in TABLE 7 had occurred in flight rather than in static firing, the overall flight performance of Saturn 1/1B would have been 66% successful

instead of 100%, and would have had a worse development characteristic than that of the non-NASA programs as can be seen from FIGURE 1.

Table 1

FLIGHT VEHICLE LAUNCH AND STATIC FIRING PROGRAM

BASIS: AVAILABLE DATA

PROGRAM	LAUNCH PERIOD	NO. FLIGHTS	NO. STATIC FIRED	TYPE STATIC FIRING
REDSTONE	8/53 - 7/61	75	57	FULL DURATION
THOR	1/57 - 9/64	179	14	FULL DURATION
JUPITER	3/57 - 5/61	39	31	FULL DURATION
ATLAS	6/57 - 9/64	223	38	SHORT DURATION
TITAN I	8/59 - 5/63	55	38	SHORT DURATION
TITAN II	3/62 - 11/64	39	12	SHORT DURATION
GEMINI	4/64 - 11/66	14	0	NONE
TITAN III	9/64 - 12/67	23	2	SHORT DURATION
SATURN I/IB	11/61 - 10/68	15	15	FULL DURATION
SATURN V	11/67 -	2	2	FULL DURATION

RHSV000745.01

Table 2

SUMMARY MILITARY MISSILE FLIGHTS

(FROM 8/53 THROUGH 12/67)

PROGRAM (PERIOD)	NO. FLIGHTS	AVG YR FLIGHT RATE	FAILURES				
			ALL CAUSES	(% TOTAL)	MISSILE* CAUSED	(% FAILURES)	LAUNCH ORDER*
REDSTONE (8/53 - 7/61)	75	9.6	21	(28)	2	(10)	3, 10
THOR (1/57 - 9/64)	179	25.0	49	(27)	8	(16)	1, 3, 6, 11, 13, 14, 20, 22, 98
JUPITER (3/57 - 5/61)	39	9.4	14	(36)	4	(28)	6, 11, 22, 25
ATLAS (6/57 - 9/64)	223	30.8	78	(35)	25	(32)	1, 2, 5, 18, 21, 23, 24, 29, 47 48, 50, 57, 58, 67, 68, 86, 95, 118, 119, 123, 156, 172, 193, 199, 211
TITAN I (8/59 - 5/63)	55	22.0	22	(40)	8	(36)	3, 16, 17, 20, 29, 37, 41, 55
TITAN II (3/62 - 11/64)	39	17	11	(28)	3	(27)	15, 17, 20
TITAN III	23	10	6	(26)	2	(33)	18, 19

RHSV000746.02

* Only those flight failures directly attributable to missile hardware systems which should have been caught by a full duration static firing. All failures resulted in destruction of the missile, either by itself or by Range Safety. Erroneous command destructs and failures attributable to missile systems not possible to be tested by static firing were specifically excluded from these data.

○ Had been static fired prior to flight.

Table 3

SUMMARY OF FIRST TWENTY-FIVE FLIGHTS

X = YES, F = FLIGHT FAILURE*, SF = HAD STATIC FIRING OF SOME KIND

FIRING ORDER	SAT I/IB		SAT V		REDSTONE		THOR		JUPITER		ATLAS		TITAN I		TITAN II		GEMINI		TITAN III	
	F	SF	F	SF	F	SF	F	SF	F	SF	F	SF	F	SF	F	SF	F	SF	F	SF
1		X		X		X	X	X		X	X	X		X		X				
2		X		X		X		X		X	X	X		X		X	X			
3		X			X	X	X	X		X		X	X	X						X
4		X				X		X		X		X		X		X				
5		X				X		X		X	X	X		X		X				
6		X				X	X	X		X		X		X		X				X
7		X				X		X	X	X		X		X						
8		X				X		X		X		X		X			X			
9		X				X		X		X		X		X						
10		X			X	X		X		X		X		X						
11		X				X	X		X	X		X		X						
12		X				X		X		X		X		X						
13		X				X	X	X		X		X		X		X				
14		X				X	X			X		X		X		X				
15		X				X				X		X		X	X	X				
16						X				X		X	X	X		X				
17						X				X		X	X	X	X	X				
18						X				X	X	X		X						
19										X		X		X					X	
20						X	X			X		X	X	X					X	
21						X				X	X			X						
22						X	X			X				X						
23						X				X	X	X		X						
24						X				X	X	X		X						
25						X				X	X	X		X						
TOTALS	0	15	0	2	2	24	8	12	2	25	8	23	4	25	2	10	2	0	2	2
# FLTS	15		2		25		25		25		25		25		25		14		23	

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* Only those flight failures directly attributable to missile hardware systems which should have been caught by a full duration static firing. All failures resulted in destruction of the missile, either by itself or by Range Safety. Erroneous command destructs and failures attributable to missile systems not possible to be tested by static firing were specifically excluded from these data.

Table 4

SUMMARY OF SATURN STAGE STATIC FIRING EXPERIENCES

AS OF DECEMBER 1, 1968

VEHICLE AND STAGE		STAGES STATIC FIRED TO DATE	NO. OF STATIC FIRINGS	NO. OF FLIGHTS	FLIGHT STAGE STATIC FIRING FAILURES					
					LAUNCH ABORT (1)		MISSION LOSS (2)		OTHER CRITICAL (4)	
					NO.	SERIAL NO. (3)	NO.	SERIAL NO. (3)	NO.	SERIAL NO. (3)
	S-I/IB	22	56	15	3	10, 12, 14	2	8, 18	2	5, 21
I & IB	S-IV	6	8	6	N/A	-	1	2	0	None
	S-IVB	9	16	3	N/A	-	2	2, 3	1	8
	S-IC	7	8	2	1	2	0	None	0	None
V	S-II	6	12	2	N/A	-	1	1	4	1
	S-IVB	8	10	2	N/A	-	0	None	1	11
TOTAL		58	110	30	4		6		8	

NOTE: (1) Occurred during first firing attempt between ignition and launch commit or would have caused de-tanking at launch.

(2) Occurred during first firing attempt or accomplishment, which would have resulted in mission loss.

(3) Chronological order No. in which fired, not stage identification – All S-IVB grouped together for this purpose since hardware is common design.

(4) Identified as the result of static firing operations which could have caused mission loss.

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Table 5

FREQUENCY OF UNPLANNED STATIC FIRING EXPERIENCES

BASIS: ALL SATURN STAGES EQUAL

SEQUENTIAL SERIAL NO.	NO. STAGES	LAUNCH ABORT	MISSION LOSS	OTHER CRITICAL	TOTAL	(%) TOTAL NO. STAGES
1	6		1	4	5	83
2	6	1	2		3	50
3	6		1		1	16
4	6					0
5	6			1	1	16
6	6					0
7	3					0
8	2		1	1	2	100
9	2					0
10	2	1			1	50
11	2			1	1	50
12	2	1			1	50
13	2					0
14	2	1			1	50
15	2					0
16	2					0
17	2					0
18	1		1		1	100
19	1					0
20	1					0
21	1			1	1	100
22	1					0
TOTAL	58	4	6	8	18	31

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TABLE 6

LAUNCH ABORTS AVOIDED

S-1-10	First static firing erroneously cut-off because of poor thrust OK switch setting. (DESIGN)
S-1B-2	Same as S-1-10 (DESIGN)
S-1D-4	Hydraulic accumulator unable to maintain pressure because of internal leakage. Had performed satisfactorily up to 10 minutes prior to ignition on first static firing countdown. (QC)
S-1C-2	LOX pump seal purge regulator on stage failed during terminal countdown for first (and only) firing; required replacement before firing. (UNKNOWN)

TABLE 7

MISSION LOSSES AVOIDED

S-1-8	6	Engine turbine bearing retainer came loose during first short-duration firing damaging turbine, resulting in thrust decrease. (DESIGN)
S-1B-8	18	Engine turbine blade failure during first short duration, caused severe damage to turbine and resulted in thrust decrease. Failure traced to improper turbine blade material used in building many engines already in field on other stages. (QC)
S-IV-6	6	Engine gimbaled hard over at ignition. Failure traced to manifold in main hydraulic pump system that had not been drilled in manufacturing process. (QC)
S-IVB-202	12	Firing 100 seconds short because fuel side propellant utilization probe indicated fuel zero right after ignition. Failure caused by loose wire (contamination) shorting out probe thus giving false reading. (QC)
S-IVB-203	13	Fuel tank pressure collapse caused insufficient pressure to engine inlet. Problem traced to design of pressurant diffuser. First experience with this small an ullage in this stage (DESIGN).

TABLE 8

OTHER CRITICAL FINDINGS

S-1-5	Fifteen to twenty, fuel, hot gas and GOX leaks occurred in the closed boattail as the result of long duration firings. (QC + DESIGN)
S-IVB-503	Destroyed by explosion during terminal countdown. If no static firing had been planned, this explosion possibly would have occurred on the Countdown Demonstration Test on the fully loaded Saturn V launch vehicle. The significance of such an occurrence is self-evident. (QC)
S-IVB-205	LOX turbine wheel found cracked through all the way around at hub of wheel. While this wheel did not fail, there is a possibility that some subsequent wheel failure could have occurred. If this had happened in flight, not only would it have been hard to definitely analyze, but it would have cost a mission and possibly lives, and many have recurred before a fix was found. (DESIGN)
S-II-1	Ignition detection probe malfunctioned causing cut-off of one engine. This is a flight safety system. Failure was traced to improper assembly. (QC)
S-II-1	High pressure LOX supply line to augmented spark igniter was found cracked after first firing. If that had happened in flight, a fire might have resulted and possibly mission loss. Failure caused redesign of this line. (DESIGN)
S-II-1	An explosion occurred in the actuator of an engine main LOX

valve. Occurrence was traced to wrong grease used in valve build-up at factory. If not found in this manner, a more drastic experience might have happened in flight where the cause would have been very hard to determine. (QC)

S-II-1	LOX pre-valve on center engine lacked proper clearance with LOX tank sump and center engine beam deflections. Pre-valve housing redesigned. In flight, interference due to acceleration could have caused loss of mission. (DESIGN)
S-IB-11	LOX pump seal failure caused fire in engine compartment. Slow reaction of launch fire detection system would have allowed launch to proceed with this catastrophic condition. (DESIGN)

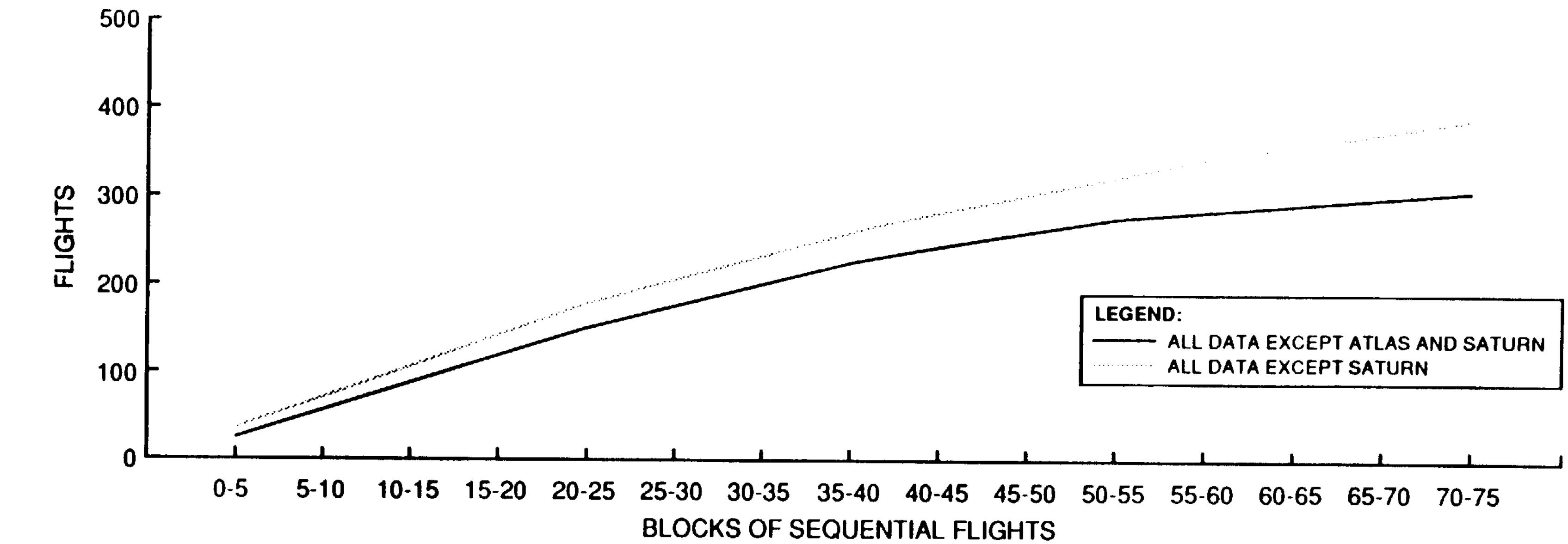

Figure 1-a. Cumulative number of flights from Aug 1953 – Dec 1967.

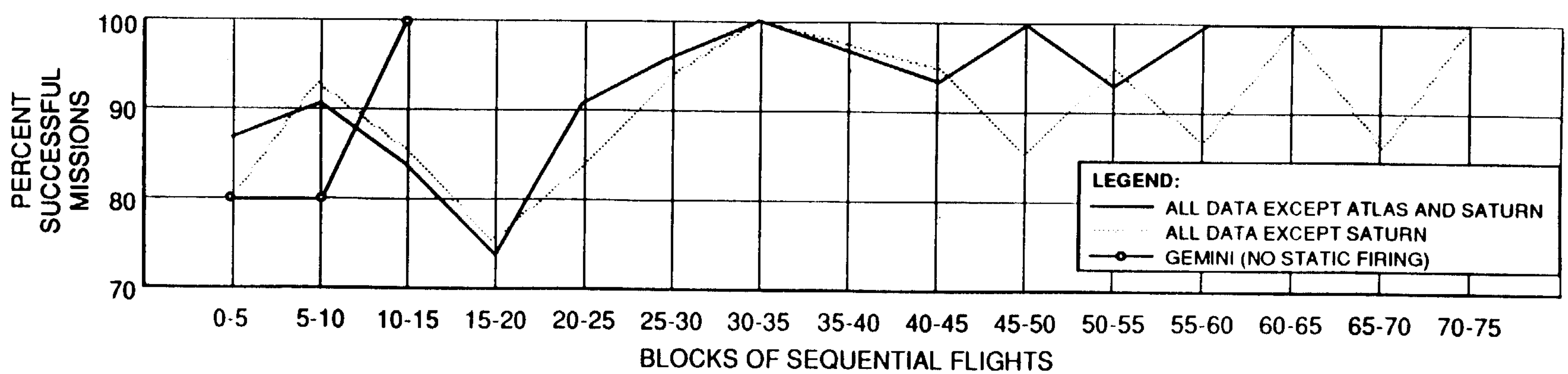

Figure 1-b. Mission success based only on failures that theoretically could have been prevented by stage acceptance firing.

NOTE: These plot scales and legends were reworked slightly from the original DHD plot of 11-27-68.

APPENDIX 11

PROPULSION COMMUNITY QUESTIONNAIRE

The questionnaire used to obtain the responses included in the body of this report is included herein for reference.

National Aeronautics and
Space Administration

John C. Stennis Space Center
Stennis Space Center, MS 39529-6000

NASA

Reply to Attn of: FA00/08MLC01

TO: Distribution

FROM: FA00/M. L. Carpenter

SUBJECT: Request for Propulsion Systems Test Information

New National Space Transportation System Vehicles are currently being planned. As the designated NASA propulsion test location, we feel it is imperative to examine lessons learned from previous propulsion system verification programs and to document your thoughts on propulsion system testing.

First, we request that you jot down (handwritten is okay) propulsion system test incidents that led to a significant design or procedural change. We are after hard evidence that shows system testing paid off by revealing problems that would have led to loss of mission or loss of human life.

Next, we request that you express your professional opinion on system verification requirements by circling the correct number of each trade criteria. This information will be used to attempt to quantify the subjective thinking of the propulsion community using the Analytical Hierarchy Process described by T. L. Saaty (McGraw Hill: NY, 1980).

In addition, a written statement that summarizes your experience, knowledge, and judgement concerning system testing would be helpful.

Your response to these requests will be incorporated into a study being performed by Rockwell's Space Division for this Center. Jim Yeager is leading the study effort to be completed in the March-April timeframe. Of course, you will be on distribution for a copy of the study report.

Your response by February 21, 1989, will be greatly appreciated.

M. L. Carpenter
M. L. Carpenter
Deputy Director
Propulsion Test Operations

QUESTIONNAIRE
PROPULSION SYSTEMS VERIFICATION REQUIREMENTS
FOR
ADVANCED PROPULSION SYSTEM

1. Please list previous propulsion system test results (or failures) that led to significant design or procedural changes. (Test site, test article, date, test number, description of results or incidents, etc.) Please limit to Saturn, Titan, Shuttle or foreign country programs of this vintage.

2. Please summarize your opinion of the role of "all-up" systems testing in verification of a new propulsion system prior to first launch.

Name______________________________

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3. Assign a numerical rating of one to nine to one criterion in each pair that you consider to be more important than the other one when you are planning a propulsion system verification program for an advanced propulsion system. The scale should be used to indicate how much more important than the other that you consider the criterion you selected to be. Please circle the number on the scale that indicates the rating you wish to assign the selected criterion. The first pair in the table has been marked as an example.

For purposes of this questionnaire, the following definitions apply:

Cost - The cost of performing the system testing and analysis.

Reliability - The increased probability of propul n system success including the reduction in probability that the launch vehicle and its payload will be destroyed and/or the fleet grounded.

Schedule - The impact on the schedule caused by the test program.

MUCH MORE IMPORTANT	MORE IMPORTANT	SOMEWHAT MORE IMPORTANT	SOMEWHAT MORE IMPORTANT	MORE IMPORTANT	MUCH MORE IMPORTANT

				COST				EXAMPLE					RELIABILITY			
9	8	7	6	5	4	3	2	1	2	3	4	5	6	7	8	9

				COST									RELIABILITY			
9	8	7	6	5	4	3	2	1	2	3	4	5	6	7	8	9

				COST									SCHEDULE			
9	8	7	6	5	4	3	2	1	2	3	4	5	6	7	8	9

				RELIABILITY									SCHEDULE			
9	8	7	6	5	4	3	2	1	2	3	4	5	6	7	8	9

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4a. The criterion for this table is *COST*. Please select the one test plan alternative from each pair in the table that you consider to be the least costly. Circle the number on the scale that indicates how much less costly you consider the alternative you selected to be. The first pair of alternatives in the table has been marked as an example.

For purposes of this questionnaire the following definitions apply for propulsion system components:

Cost - The cost of performing the system testing and analysis.

Analysis Only - Components are tested on individual basis and not as a system. System verification is by analysis.

Battleship Test - Components are tested on individual basis and then installed for testing on heavyweight tanks in a flight system configuration.

Flight Configuration Test Article - Components are tested on individual basis and then installed in an actual flight configuration test article for static firing tests.

FRF Test Only - Components are tested on an individual basis then installed on a flight vehicle which is checked out in a short duration static firing on the launch pad.

Flight Test - Components are tested on an individual basis, then installed in a launch vehicle and launched on a test flight.

CRITERION = COST

MUCH MORE PREFERRED		MORE PREFERRED		SOMEWHAT MORE PREFERRED						SOMEWHAT MORE PREFERRED		MORE PREFERRED			MUCH MORE PREFERRED	
		ANALYSIS ONLY				*EXAMPLE*				*ESHIP TEST PLUS ANALYSIS*						
9	8	7	6	5	4	3	2	1	2	3	4	5	6	7	8	9

ANALYSIS ONLY									*BATTLESHIP TEST PLUS ANALYSIS*							
9	8	7	6	5	4	3	2	1	2	3	4	5	6	7	8	9

ANALYSIS ONLY									*FLIGHT CONFIGURATION TEST ARTICLE PLUS ANALYSIS*							
9	8	7	6	5	4	3	2	1	2	3	4	5	6	7	8	9

ANALYSIS ONLY									*FRF TEST ONLY PLUS ANALYSIS*							
9	8	7	6	5	4	3	2	1	2	3	4	5	6	7	8	9

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CRITERION = COST

MUCH MORE PREFERRED	MORE PREFERRED	SOMEWHAT MORE PREFERRED	SOMEWHAT MORE PREFERRED	MORE PREFERRED	MUCH MORE PREFERRED

ANALYSIS ONLY	*FLIGHT TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

BATTLESHIP TEST PLUS ANALYSIS	*FLIGHT CONFIGURATION TEST ARTICLE PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

BATTLESHIP TEST PLUS ANALYSIS	*FRF TEST ONLY PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

BATTLESHIP TEST PLUS ANALYSIS	*FLIGHT TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

FLIGHT CONFIGURATION TEST ARTICLE PLUS ANALYSIS	*FRF TEST ONLY PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

FLIGHT CONFIGURATION TEST ARTICLE PLUS ANALYSIS	*FLIGHT TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

FRF TEST ONLY PLUS ANALYSIS	*FLIGHT TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

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4b. The criterion for this table is *RELIABILITY*. Please select the one test plan alternative from each pair in the table that you consider to result in the most reliable propulsion system. Circle the number on the scale that indicates how much more reliability you consider the alternative you selected will provide. The first pair of alternatives in the table has been marked as an example.

For purposes of this questionnaire the following definitions apply for propulsion system components:

Reliability - The increased probability of propulsion system success including the reduction in probability that the launch vehicle and its payload will be destroyed and/or the fleet grounded.

Analysis Only - Components are tested on individual basis and not as a system. System verification is by analysis.

Battleship Test - Components are tested on individual basis and then installed for testing on heavyweight tanks in a flight system configuration.

Flight Configuration Test Article - Components are tested on individual basis and then installed in an actual flight configuration test article for static firing tests.

FRF Test Only - Components are tested on an individual basis then installed on a flight vehicle which is checked out in a short duration static firing on the launch pad.

Flight Test - Components are tested on an individual basis, then installed in a launch vehicle and launched on a test flight.

CRITERION = RELIABILITY

MUCH MORE PREFERRED	MORE PREFERRED	SOMEWHAT MORE PREFERRED	SOMEWHAT MORE PREFERRED	MORE PREFERRED	MUCH MORE PREFERRED

ANALYSIS ONLY	*EXAMPLE*	*ESHIP TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2	1	2 3 4 5 6 7 8 9

ANALYSIS ONLY		*BATTLESHIP TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2	1	2 3 4 5 6 7 8 9

ANALYSIS ONLY		*FLIGHT CONFIGURATION TEST ARTICLE PLUS ANALYSIS*
9 8 7 6 5 4 3 2	1	2 3 4 5 6 7 8 9

ANALYSIS ONLY		*FRF TEST ONLY PLUS ANALYSIS*
9 8 7 6 5 4 3 2	1	2 3 4 5 6 7 8 9

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CRITERION = RELIABILITY

MUCH MORE PREFERRED — MORE PREFERRED — SOMEWHAT MORE PREFERRED — SOMEWHAT MORE PREFERRED — MORE PREFERRED — MUCH MORE PREFERRED

ANALYSIS ONLY	*FLIGHT TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

BATTLESHIP TEST PLUS ANALYSIS	*FLIGHT CONFIGURATION TEST ARTICLE PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

BATTLESHIP TEST PLUS ANALYSIS	*FRF TEST ONLY PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

BATTLESHIP TEST PLUS ANALYSIS	*FLIGHT TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

FLIGHT CONFIGURATION TEST ARTICLE PLUS ANALYSIS	*FRF TEST ONLY PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

FLIGHT CONFIGURATION TEST ARTICLE PLUS ANALYSIS	*FLIGHT TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

FRF TEST ONLY PLUS ANALYSIS	*FLIGHT TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

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4c. The criterion for this table is *SCHEDULE*. Please select the one test plan alternative from each pair in the table that you consider to cause the least schedule impact. Circle the number on the scale that indicates how much less schedule impact you consider the alternative you selected to have. The first pair of alternatives in the table has been marked as an example.

For purposes of this questionnaire the following definitions apply for propulsion system components:

Schedule - The impact on the schedule caused by the test program.

Analysis Only - Components are tested on individual basis and not as a system. System verification is by analysis.

Battleship Test - Components are tested on individual basis and then installed for testing on heavyweight tanks in a flight system configuration.

Flight Configuration Test Article - Components are tested on individual basis and then installed in an actual flight configuration test article for static firing tests.

FRF Test Only - Components are tested on an individual basis then installed on a flight vehicle which is checked out in a short duration static firing on the launch pad.

Flight Test - Components are tested on an individual basis, then installed in a launch vehicle and launched on a test flight.

CRITERION = SCHEDULE

MUCH MORE PREFERRED		MORE PREFERRED		SOMEWHAT MORE PREFERRED		SOMEWHAT MORE PREFERRED		MORE PREFERRED		MUCH MORE PREFERRED

ANALYSIS ONLY								EXAMPLE								ESHIP TEST PLUS ANALYSIS
9	8	7	6	5	4	3	2	1	2	3	4	5	6	7	8	9

ANALYSIS ONLY									BATTLESHIP TEST PLUS ANALYSIS							
9	8	7	6	5	4	3	2	1	2	3	4	5	6	7	8	9

ANALYSIS ONLY									FLIGHT CONFIGURATION TEST ARTICLE PLUS ANALYSIS							
9	8	7	6	5	4	3	2	1	2	3	4	5	6	7	8	9

ANALYSIS ONLY									FRF TEST ONLY PLUS ANALYSIS							
9	8	7	6	5	4	3	2	1	2	3	4	5	6	7	8	9

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CRITERION = SCHEDULE

MUCH MORE PREFERRED	MORE PREFERRED	SOMEWHAT MORE PREFERRED	SOMEWHAT MORE PREFERRED	MORE PREFERRED	MUCH MORE PREFERRED

ANALYSIS ONLY	*FLIGHT TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

BATTLESHIP TEST PLUS ANALYSIS	*FLIGHT CONFIGURATION TEST ARTICLE PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

BATTLESHIP TEST PLUS ANALYSIS	*FRF TEST ONLY PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

BATTLESHIP TEST PLUS ANALYSIS	*FLIGHT TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

FLIGHT CONFIGURATION TEST ARTICLE PLUS ANALYSIS	*FRF TEST ONLY PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

FLIGHT CONFIGURATION TEST ARTICLE PLUS ANALYSIS	*FLIGHT TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

FRF TEST ONLY PLUS ANALYSIS	*FLIGHT TEST PLUS ANALYSIS*
9 8 7 6 5 4 3 2 1	2 3 4 5 6 7 8 9

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